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MANNED SPACECRAFT CENTER

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

APOLLO SPACECRAFT PROJECT

STATUS REPORT NO. 1

FOR

PERIOD ENDING SEPTEMBER 30, 1962

By Manned Spacecraft Center

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FOREWORD

This report is the first in a series of reports on the status of the APOLLO Spacecraft Project for the Manned Lunar Landing Program.

This first report describes the functions and requirements of the spacecraft modules and systems as well as their current development status. Subsequent reports will cover only the progress accomplished during the reporting period.

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SUMMARY

The spacecraft and launch vehicle, being developed by the Manned Spacecraft Center and Marshall Space Flight Center, respectively, comprises the APOLLO Space Vehicle (fig. 1). The configuration of the APOLLO spacecraft is shown in figure 2.

The spacecraft is composed of separable modules including a Command Module which houses the crew from the earth to the vicinity of the moon and the return to the earth; a Service Module which contains propulsion for midcourse, lunar orbit and deorbit velocity changes, and other stores and systems; and a Lunar Excursion Module which separates from the Command and Service Modules when in lunar orbit and descends to the lunar surface for manned exploration.

The Saturn C-5 will be the basic launch vehicle for lunar missions. The C-5 consists of three stages, the S-IC, S-II, and S-IVB. The S-IC uses LOX-RP-1 propellants and five F-1 engines. The S-II stage uses LOX-LH_2 propellants and five J-2 engines. The S-IVB stage uses LOX-LH_2 propellants and one J-2 engine.

Major accomplishments since the inception of the project are:

a. The APOLLO Spacecraft Project Office was established and organized within the Manned Spacecraft Center.

b. Technical guidelines for the spacecraft and the mission were formulated.

c. Lunar mission feasibility studies were conducted both inhouse and by three industrial contractors.

d. A contract was awarded to the MIT-Instrumentation Laboratory for the development of the Navigation and Guidance System.

e. North American Aviation, Inc., Space and Information Systems Division, was selected as principal spacecraft contractor with detailed responsibility for the development of the Command and Service Modules.

f. A contract was awarded to General Dynamics-Convair for the development of the Little Joe-II Test Launch Vehicle.

g. AC Spark Plug Company, Raytheon Company, and Kollsman Instrument Corporation were awarded contracts for the development of portions of the Navigation and Guidance System.

h. Detailed studies were conducted of the lunar orbit rendezvous mission mode and the Lunar Excursion Module.

i. Evaluation of the industrial proposals for the Lunar Excursion Module development was completed.

j. Evaluation of the industrial proposals for the Mission Control Center Computer was completed and IBM was selected for negotiations.

k. Design studies and detail engineering have defined the primary features of the Command Module and Service Module configurations, structures, and functional systems. Approximately one-half of the key drawings for the Command Module and Service Module have been released for manufacturing. Procurement specifications for most of the long lead-time items have also been released. Except for the computer, the Navigation and Guidance System is nearing design freeze.

l. Definitization of the North American Aviation spacecraft contract is in progress.

Significant accomplishments and design changes of the past 3 months are shown below:

a. Mock-ups of the Command Module interior arrangement and the Navigation and Guidance System were completed.

b. Two ground test Command Module boilerplates were completed and tested with good results.

c The stabilization subsystem of the Launch Escape System was changed to a single kicker-rocket subsystem having but one functional mode.

d. Windows for direct external viewing have been added to the Command Module.

e. The 7.0-psia 50-50 oxygen-nitrogen cabin atmosphere was changed to 5.0-psia pure oxygen.

f. The man-sized airlock was deleted.

g. The landing mode is semi-passive. No orientation control will be required, and no Command Module impact attenuation system will be employed.

h. The Command Module Reaction Control System thrusters have been changed to match those used in Project GEMINI. Propellants in both the Command and Service Modules Reaction Control Systems have been changed to MMH-N_2O_4.

i. The Reaction Control System engine expansion ratio has been developed to a ratio of 60:1.

j. The Service Module Propulsion System tanks are now sized to hold 45,000 pounds of useful propellant. This increased capacity will allow greater operational flexibility if greater launch vehicle performance and/or lower spacecraft weight is realized.

k. The Service Module length has been reduced from 155 inches to 140 inches.

l. The Environment Control System and Electrical Propulsion System heat rejection radiators were reduced in size and made nondeployable.

m. Provisions for installation of a paraglider are no longer required.

n. Personal parachutes are no longer required. Individual survival gear has been deleted in favor of combined survival gear.

o. It was established that there would be four crew-duty stations in the Command Module.

p. The mounting arrangement for the optical and inertial measurement units of the Navigation and Guidance System was changed to a fixed-base mount. This change decreases the system power requirements since the former extendable arrangment required power to reposition the units.

MISSION PLAN

The mission objective of Project APOLLO is the landing of men on the lunar surface, limited observation and exploration of the lunar surface by the crew in the landing area, and return of the crew to the earth. The APOLLO development plan envisages the qualification of the spacecraft for lunar missions by a series of increasingly complex missions including suborbital and earth orbital missions.

A nominal description of the lunar landing mission plan (fig. 3) is:

a. The spacecraft will be launched from Cape Canaveral into a 100-nautical-mile circular parking orbit by a Saturn C-5 launch vehicle.

b. After approximately $1\frac{1}{2}$ orbits, the S-IVB stage of the Saturn C-5 launch vehicle will be restarted to inject the spacecraft into a translunar[1] trajectory.

c. During the nearly three days required to reach the moon, the crew will navigate the spacecraft and make several midcourse corrections using the Service Module Propulsion System.

d. As the spacecraft reaches the point of closest approach, the Service Module is restarted to slow the spacecraft into an 80-nautical-mile circular orbit about the moon.

e. After the orbit has been established, two men will transfer to the Lunar Excursion Module. The Lunar Excursion Module then separates from the Command Module and the crew prepares for the lunar landing.

f. The Lunar Excursion Module will then propel itself into an elliptic lunar orbit with a 50,000-foot pericynthion which will allow an inspection of the landing area.

g. After one or two orbits, the Lunar Excursion Module crew will have selected the exact landing site. The crew will then restart the descent engine which will provide thrust to make the approach, flare, hover, and touchdown.

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[1] Editors note: The use of the word "translunar" in this report refers to that phase of the overall lunar mission during which the spacecraft is in transit from the earth orbit to the lunar orbit.

h. While on the lunar surface (approximately 1 day), the crew will prepare for launch, collect soil samples, and conduct various scientific experiments.

i. After a systems checkout, the Lunar Excursion Module will be launched, using its ascent engine, into an elliptical orbit having an 80-nautical mile apocynthion, and there rendezvous with the Command and Service Modules.

j. After rendezvous and docking, the crew and cargo will transfer to the Command Module and preparations for the return trip will begin. The Lunar Excursion Module will then be jettisoned.

k. At the proper time, the Service Module Propulsion System will be started, propelling the Command and Service Module into the transearth[2] trajectory.

l. Approximately 3 days later, the Command Module will separate from the Service Module, assume reentry attitude, and reenter the earth's atmosphere.

m. During reentry, the landing range will be controlled by modulation of the lift vector through roll control; thereby enabling the Command Module to reach one of several primary landing sites.

n. At an altitude of approximately 25,000 feet, the drogue parachute will be deployed, and the three main parachutes will then be deployed at an altitude of 15,000 feet.

A detailed analysis of the various maneuvers required throughout the mission has been completed to determine the total incremental velocity required for the Service Module engine and for the Lunar Excursion Module engine.

Including adverse geometric factors and a 10-percent control allowance, the Service Module will provide a total velocity increment of 3,880 feet per second during the translunar trajectory phase of the mission and 4,800 feet per second during the transearth trajectory phase. This is equivalent to a total of 40,300 pounds of propellants for a 90,000-pound (escape weight) spacecraft if the Lunar Excursion Module is jettisoned prior to returning to earth.

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[2] Editor's note: The use of the term "transearth" in this report refers to that phase of the overall lunar mission during which the spacecraft is in transit from the lunar orbit to the earth orbit.

The Lunar Excursion Module will require a velocity increment of approximately 7,740 feet per second for braking from lunar orbit to lunar landing, and a velocity increment of 6,880 feet per second for the phase from lunar launch to docking with the Command Module. These velocities are equivalent to a total of 18,000 pounds of propellants, if the Lunar Excursion Module weight is 26,000 pounds at separation from the Command Module, and the lunar launch weight is 7,200 pounds. This analysis shows that the propellants required for a typical lunar-landing mission will be some 58,500 pounds or approximately two-thirds of a 90,000-pound spacecraft. A summary of the ΔV capabilities of the spacecraft for the various mission phases is presented in table 1. Detail studies to define the mission plan further are continuing both in the MSC and by the contractors.

TABLE I

SPACECRAFT	MISSION PHASE	VELOCITY INCREMENT (FT/SEC)
SERVICE MODULE	TRANSLUNAR	
	Midcourse	300
	Coplanar Retro to 80 NM Orbit	3130
	6° Plane Change	100
	10 percent Control Allowance	353
	TOTAL ΔV_1 (LEM & CM attached)	3883
	TRANSEARTH	
	Lunar Excursion Module Pickup	455
	Transearth Injection	3610
	Midcourse Correction	300
	10 percent Control Allowance	436
	TOTAL ΔV_2 (only CM attached)	4801
LUNAR EXCURSION MODULE	LUNAR ORBIT TO LUNAR SURFACE	
	Separation from CM	5
	Injection into Equal Period Orbit (80 NM to 50,000 ft.)	373
	Descent to 1,000 ft.	5961
	Hover, Translate & Touchdown	700
	10 percent Control Allowance	704
	SUB-TOTAL ΔV	7743
	LUNAR SURFACE TO LUNAR ORBIT	
	Launch to 50,000 ft. (T/W=0.4)	5885
	2° Plane Change During Launch	75
	Rendezvous from 50,000 ft. to 80 NM	196
	Abort Capability	100
	10 percent Control Allowance	626
	SUB-TOTAL ΔV	6882
	TOTAL LEM ΔV	14610

SPACECRAFT AND ADAPTER DESIGN AND DEVELOPMENT

The APOLLO Spacecraft is composed of a Command Module, Service Module, and Lunar Excursion Module. An adapter provides the attachment between the launch vehicle and the spacecraft. The Lunar Excursion Module is housed in this adapter during launch.

The prime APOLLO Spacecraft development contract has been awarded to North American Aviation, Inc., Spacecraft and Information Systems Division, for development of the Command and Service Modules, adapter, associated ground support equipment, and spacecraft integration. Massachusetts Institute of Technology, Instrumentation Laboratory, has been awarded a contract for the spacecraft Navigation and Guidance System for the Command and Service Modules and the Lunar Excursion Module. The Lunar Excursion Module contractor has not been selected. The MSC with the support of various contractors will provide research and development instrumentation, scientific instrumentation, and personal equipment for the crew.

COMMAND AND SERVICE MODULES

The Command Module is the Space Vehicle command center from which all crew-initiated control functions are exercised during launch, translunar, transearth, earth reentry, and landing phases of the mission.

The Service Module is unmanned and contains propulsion systems for midcourse correction, entry and exit into lunar orbit, and for lunar-orbit rendezvous as a backup to the Lunar Excursion Module propulsion. The Service Module is non-recoverable and will be jettisoned prior to earth reentry.

The prime contract was awarded to North American Aviation, Inc., Space and Information Systems Division for design and development of the APOLLO Spacecraft in December 1961. Contract definitization is currently in progress.

Design studies and detail engineering drawings have defined primary features of the Command Module and Service Module configurations, structures, and functional systems. Approximately one-half of the key drawings for the Command Module and Service Module have been released for manufacturing, and procurement specifications for most of the long lead-time items have been released.

Description and status of the principal systems of the Command and Service Modules are discussed in the following paragraphs.

Launch Escape System

The Launch Escape System (LES) provides an abort capability for the Command Module during the time prior to launch until approximately 10 seconds after second-stage ignition. During this period in the launch trajectory, three critical conditions determine performance design criteria: (1) pad abort, (2) maximum dynamic pressure (q) abort, and (3) high altitude abort. Sufficient altitude must be obtained from a pad abort to allow reliable deployment of the landing system and sufficient range to clear the launching area. The minimum requirements have been established as an altitude of 4,000 feet and a range of 3,000 feet at the apogee. At maximum q, the thrust level must be high enough to overcome aerodynamic drag and provide sufficient acceleration to carry the Command Module a safe distance from a non-thrusting launch vehicle and remain within crew acceleration tolerances. The LES consists of an open truss tower, escape rocket, pitch control rocket, and tower jettison rocket.

The escape rocket is 26 inches in diameter, 186 inches long, and contains a case-bonded, internal-burning, 8-point-star propellant grain consisting of polysulfide fuel binder and ammonium perchlorate oxidizer. The nozzle assembly consists of 4 nozzles, canted 35° to the motor centerline. The motor is ignited with a head-end pyrogen-type igniter containing two exploding bridge wire (EBW) initiators. The thrust level is 155,000 pounds average during 1.5 seconds of burning at 70° F and 36,000 foot altitude and is directed along a line canted 2°45' from the motor centerline in the pitch plane. The burning time is 3.5 seconds and total time is 8 seconds.

The pitch control motor provides the launch-escape configuration with a positive pitching moment to control the abort trajectory and is ignited about 70 milliseconds after escape motor ignition to allow time for escape motor thrust buildup. The design of the pitch control motor is such that, during manufacturing, the performance can be varied between 1,200 lb-sec and 3,000 lb-sec of impulse by changing the propellant loading and nozzle sizing. This feature will allow precise abort trajectory control based on the results of preliminary flight tests. The pitch control motor is 9 inches in diameter and 22 inches long. The polysulfide-ammonium perchlorate internal burning propellant grain has a nominal burning time of 0.5 seconds.

The contract for the escape rocket was awarded to Lockheed Propulsion Company in February 1962. Initial requirements were for a 200,000-pound-thrust rocket motor with an active thrust vector control subsystem. After extensive study of both a liquid secondary injection and hinged nozzle control subsystem, Lockheed was directed to remove the control subsystem. A letter contract change was subsequently made with Lockheed to develop and manufacture a pitch-control motor to replace the thrust-vector-control subsystem. In conjunction with the use of the pitch-control motor, the escape-motor thrust was reduced to 155,000 pounds, thus requiring a redesign of the propellant grain and aft closure. At the present time, all designs of the escape and pitch control motors are complete and all drawings have been released for manufacture. Preliminary ballistic motor and igniter testing has begun. The first inert escape motor has been completed. The first static firings of the escape and pitch-control motors are scheduled for early December 1962.

The tower-jettison motor provides the capability for jettisoning the LES during a normal mission or following a launch abort. In the normal mission, the escape motor serves as backup to the tower jettison motor for LES separation. The jettison motor is 26 inches in diameter and 47 inches long. It has two nozzles canted 30° to the motor centerline. The propellant grain is a polysulfide/ammonium perchlorate, case-bonded, internal-burning star configuration. The motor uses an aft mounted pyrogen-type igniter having two EBW initiators. The resultant thrust of 33,000 pounds, for a burning time of 1 second, is directed along a line canted 2.5° from the motor centerline.

The contract for the tower-jettison motor was awarded to Thiokol Chemical Corporation in April 1962. No major design redirections have been made since the inception of the program. All detailed designs have been completed and all drawings have been released for manufacture. Preliminary nozzle evaluation, ballistic motor, and igniter tests have been completed and the test results have confirmed design calculations. The first static firing is scheduled for late October 1962.

Command Module Structural System

The Command Module structure consists of two basic elements: the inner structure and the heat shield.

The function of the inner structure is to counteract the primary loads such as pressure, body loads, parachute opening loads, landing loads, and the inertia loads of equipment. The inner structure is fabricated from 2014 aluminum-alloy bonded-honeycomb-sandwich construction.

The body loads and the pressure loads are carried by the sandwich skins and the locally applied loads, such as parachute opening loads, are reacted and distributed into the sandwich skins by the longerons and gussets. The structural design release of the Command Module is 65 percent complete with 100 percent release scheduled for January 1963.

The base heat shield is a composite structure made up of a charring ablator, a stainless-steel honeycomb sandwich and a layer of insulation. The contract for ablation material development was awarded to AVCO. The ablator is AVCOAT 5026 fabricated in tiles. A combination of bonding and mechanical fasteners will be used to secure the tiles to the PH15-7 Mo stainless steel sandwich. The ablator formulation is being refined for performance and fabrication. The final selection will be made in the near future.

Service Module Structural System

The Service Module structure is made up of six cylindrical aluminum alloy 7075 honeycomb-sandwich panels, six deep radial longerons and a honeycomb-sandwich aft bulkhead. Changes in length have been incorporated to accommodate variations in propellant loading. The Service Module structural design is less than 10 percent released; however, 65 percent release is due by mid-December 1962.

Crew Equipment

Crew equipment includes an instrument panel for manual control and observation of the progress of the flight; a couch and restraint system for protection from the effects of acceleration, impact, and vibration, and for restraint in weightlessness; space suits for decompression protection and for extra-vehicular operations; food and water; sanitation equipment; emergency equipment including radiation dosimeters, medical instruments, first aid equipment, and post-landing survival equipment.

Layout and location of the instrument panel will be primarily determined by crew arrangement and function. North American Aviation, Inc. has established an active Controls-Displays Working Group consisting of electronics, flight-dynamics, and human-factors specialists. This group made its first status report in August 1962. The current tentative seating arrangement calls for four primary duty stations:

Control Station (left side of main panel)

Center Station (center of main panel)

Systems Management Station (right side of main panel)

Navigation Station (at lower equipment; near feet of center crew man)

The center station is inactive and the couch is removed during most flight phases. Mock-ups of a preliminary instrument panel have been fabricated.

Initial impact tests, using boilerplate command modules, have shown the impact acceleration of 20 g's to be attenuated to 5 g's at the couch. Personal parachutes were replaced with combined survival gear as a result of the elimination of the requirement for bailout escape. The parachutes were to be stored under the couch for ready access and required a rigid couch structure. This requirement was a primary factor in the current design of the couch. Deletion of this requirement will result in a redesign of the couch which is now underway. Comparative studies are being conducted using the present basic contoured couch, which has a surface material that forms to the individual crew member's shape during use, and a net-couch. As a connecting member to the couch, the restraint harness is also subject to redesign. Also under study is the use of the couch to obtain the necessary center of gravity change for reentry through repositioning of the crew members.

A description and status of the space suit is presented in the section entitled Space Suit System.

Freeze-dried food, which will be used in the Project GEMINI missions, will also be used for Project APOLLO. Forty-two pounds of food will be provided for a 14-day lunar-landing mission. Potable water will be supplied by the fuel cells and processed by the Environmental Control System. A 1-day water supply of 6 pounds-per-man will be provided at launch as emergency ration, if needed before the fuel cell is fully operative.

A wide range of methods for both personal- and litter-waste management is under study to determine the optimum method. Water for sanitation is in ready supply from the fuel cell.

Radiation dosimeters, medical instruments, and first aid equipment requirements have been determined in detail. Approximately eight medical instruments which may be used for minor operations, plus various drugs, will be included.

A three-man liferaft will replace the individual liferafts as a major component of the post-landing survival equipment. A prototype of this raft has been developed. Other survival equipment includes small articles, such as those used in Project MERCURY and which are presently planned for use in Project GEMINI. This equipment will provide for nourishment, first aid, and protection prior to recovery.

Environmental Control System

The Environmental Control System (ECS) for the Command and Service Modules will provide a 5 psia pure-oxygen shirt-sleeve atmosphere for the crew; conditioned-oxygen atmosphere at 3.5 psia to the pressure suits in event of cabin decompression; thermal control of onboard critical equipment; charging of self-contained extra-vehicular pressure suit support systems; and water for crew consumption and heat-transfer operations. Environmental control is accomplished with two gas circulation loops, a gas supply system, and a thermal control system. The systems maintain the cabin pressure at 5 psia, relative humidity between 40 and 70 percent, partial pressure CO_2 at maximum 7.6 mm Hg, and a temperature of 70° to 80° F during normal conditions.

The Environmental Control System contract was awarded to AiResearch Manufacturing Company.

The heart of the ECS is a package which is 31 inches high by 36 inches wide by $17\frac{1}{2}$ inches deep, that weighs approximately 160 pounds, and that can be completely removed from the spacecraft by breaking only eleven connections. The system is easily accessible to the crew and most of the vital functions are actuated or backed up by manually operated, direct mechanical linkages.

A 20-pound, 2,000-cubic-inch saving has been effected by elimination of the sponge-type water separator; current plans are to replace it with a wick separator (based on capillary action) or a centrifugal separator. The use of a centrifugal separator is still under study; since the weight, volume, and power requirements for the wick and centrifugal separator are approximately equal.

Detailed plans for breadboard testing as well as designs of test instrumentation and facilities have been prepared through the prelaunch mission phase.

A liquid coolant system using water-glycol is being evaluated from the standpoint of weight reduction, operation during various mission modes, and inflight servicing.

Guidance and Navigation System

The Guidance and Navigation System provides for the guidance of the APOLLO spacecraft during powered phases of its flight, which include midcourse velocity corrections, injection into lunar orbit, injection into transearth trajectory phase, and reentry. The navigational portion of the system will provide position and velocity information during the midcourse, lunar-orbit, and reentry operations.

The major components of the Guidance and Navigation System are the inertial measurement unit, the optical measurement unit, the guidance computer, the coupling display unit, associated electronic packages, and controls and information displays.

The inertial measurement unit (IMU) provides control of the spacecraft during powered flight and reentry and determines the amount of velocity change actually achieved during thrusting periods.

The optical measurement unit (OMU) consists primarily of a space sextant and scanning telescope. The space sextant is used to measure precision angles between landmarks on either the earth or the moon and selected stars. The scanning telescope is used as a finder for the sextant, for alignment purposes when setting up the IMU, and for landmark sightings while in earth or lunar orbit.

The guidance computer is the heart of the guidance and navigation system, assimilating information from the navigation system and calculating orders for the guidance system.

The computer uses the measurements from the IMU and OMU in calculating the spacecraft position and velocity. This information is used to make corrections to the spacecraft trajectory during midcourse flight. The guidance computer is a general-purpose unit having about a 12,000-word fixed-storage and about a 2,000-word eraseable storage. The computer will occupy about 1.7 cubic feet of space and will weigh about 100 pounds.

The coupling display units, which are used primarily as analog-to-digital converters, are the means by which the computer will issue commands to the guidance portion of the system.

The Power and Servo Assembly, an integrated assembly of electronic power supplies and analog control electronics, provides power and control instrumentation for all navigation and guidance subsystems.

MIT Instrumentation Laboratory is responsible for the design and analysis of the APOLLO Spacecraft Guidance and Navigation System. A. C. Spark Plug Company, Milwaukee, has a letter contract for the manufacture of the IMU, coupling display unit, and certain portions of the other displays; for systems assembly and test; and for major ground-support equipment and the integration of other navigation contractor ground-support equipment. A. C. Spark Plug Company has separate contracts for the Power and Servo Assembly and for producing the gyros used in the IMU. Sperry Gyroscope Company has a contract for the accelerometers which are used in the IMU. Kollsman Instrument Company has a letter contract for the scanning telescope, the sextant, and certain other display units. Raytheon Company, Sudbury, Massachusetts, has a letter contract for the guidance computer.

All of the letter contracts are in the process of being formalized into definitive contracts.

The majority of the components in the Guidance and Navigation System are nearing a design freeze; however, certain items such as the fixed storage circuits for the computer and certain of the computer programs will not be started until mission plans and more firm requirements are available.

Sperry Gyroscope is now manufacturing the accelerometers and will deliver the instruments to Massachusetts Institute of Technology for evaluation and incorporation into prototype systems as required. A. C. Spark Plug Company is also manufacturing the gyros and should have a gyro on test in January 1963.

Stabilization and Control System

The spacecraft Stabilization and Control System accepts attitude and steering information from the guidance system and provides attitude control and stabilization of the spacecraft, backup attitude reference, manual control capability, and means for backup navigation and guidance of the spacecraft. The system displays attitude, attitude rate, attitude errors, velocity increment and reentry deceleration. This system also provides lateral control of the spacecraft through a three-axis control stick as well as control of the Service Module Propulsion System.

The Stabilization and Control System consists of the following major subassemblies: body-mounted attitude reference gyros; mounted rate gyros; a longitudinal accelerometer; electronic assemblies for control of the reaction control jets and the Service Module Propulsion System; and astronaut controls and displays. The attitude reference gyros are three body-mounted single-axis-floated gyros. They will provide attitude reference information during prolonged coasting periods when the guidance system IMU is shut down. They also provide attitude reference backup in the event of IMU malfunction. The rate gyro package consists of three rate gyros which provide angular rate information for stabilization and display. Both the rate gyros and the attitude reference gyros may be replaced inflight without using precision equipment. The longitudinal accelerometer is a backup to the guidance system IMU during thrusting periods and during reentry. The control system electronic package contains all the necessary amplifiers, driver circuits, and gain and switching logic required for the proper operation of the Service Module Propulsion System and the Reaction Control System. The astronaut controls will consist of two controllers: one, for spacecraft angular motion in three axes and the other for spacecraft translational motion also in three axes. Exact physical form for these controllers has not been determined although a number of study models have been made. Other astronaut controls are attitude reference selectors, control system deadband and gain controls, and attitude display controls. The astronaut displays consist primarily of attitude as displayed on an eight-ball attitude indicator, attitude errors, and attitude rates. In addition, incremental velocity and reentry deceleration information is displayed. Gimbal angles of the Service Module Propulsion System engine are also displayed.

The spacecraft Stabilization and Control System is a part of North American Aviation's overall contract, and this system has been subcontracted to Minneapolis-Honeywell. At present, North American Aviation is operating under a letter contract and their subcontract to Minneapolis-Honeywell is presently under negotiation.

The Stabilization and Control System conceptual studies and designs have been completed.

Reaction Control System

The Reaction Control System (RCS) provides the impulse for attitude control and stabilization of the Command and Service Modules. The system is ablation-cooled, pulse modulated, pressure fed, and uses earth-storable hypergolic fuel. Fuel tanks are of the positive-expulsion type.

The Command Module dual RCS provides three-axis control prior to the development of aerodynamic moments, roll control during reentry and landing, pitch and yaw rate damping during reentry and deployment of the landing system, as well as tumbling arrest in the event of abort. This system will be used only after separation of the Command Module from the Service Module.

The Service Module quadruple modular RCS provides impulse for attitude control and stabilization of the Space Vehicle in all phases of flight, except during periods that other propulsion systems are active, and during earth-parking orbit. The system also provides a translational capability for minor midcourse corrections, terminal rendezvous and docking, as well as ullage accelerations for the Service Module Propulsion System if necessary. Command and Service Module RCS engines design and performance requirements are shown in table 2.

The Contract for the Command and Service Module RCS engines was awarded to Marquardt in April. A subsequent decision was made to use the GEMINI spacecraft Orbital Attitude Maneuvering System's (OAMS) 100-pound thruster for the Command Module and, therefore, fabrication of the Command Module RCS engine will be accomplished by Rocketdyne. Use of the GEMINI spacecraft engine will require an increase in Command Module propellant system pressure, an increase in minimum pulse width, and a change in fuel from Aerozine 50 (50-50 UDMH/N_2H_4) to Monomethyl hydrazine (MMH); preliminary revised specifications have been prepared.

The plumbing configuration of the Service Module RCS was changed from a dual to a quadruple modular system. The first 100-pound engine has been completed and fired for a total of 168 seconds in eight runs with good results. Fabrication of the Command Module RCS breadboard test fixture is completed and the Service Module fixture is 75 percent complete.

TABLE 2

Performance	Requirements
Propellants	MMH Fuel, N_2O_4 oxidizer
Vacuum thrust level	100 $\pm$5 pounds
Vacuum specific impulse	300 sec (steady state)
	250 sec (pulse mode - 10 ms)
Minimum impulse bit	2 pound-sec (Command Module)
	1 pound-sec (Service Module)
Continuous operation	200 seconds (CM)
	1000 seconds (SM)
Operational cycles	9000/200 seconds (CM)
	10,000/1000 seconds (SM)
Reliability	0.99948/200 seconds (CM)
	0.994/1000 seconds (SM)

The fuel for the Command and Service Modules RCS's was changed from 50-50 UDMH/N_2H_4 to MMH. Specifications have been released for all Command and Service Modules helium system components.

Service Module Propulsion System

The Service Module Propulsion System furnishes all thrust for major velocity changes required for all translunar and transearth maneuvers, placement of the spacecraft into lunar orbit, backup mode for rendezvous of the Lunar Excursion Module with the Command and Service Modules, and transfer of the Command and Service Modules from lunar orbit to the trans-earth trajectory, as well as capability for abort after jettison of the Launch Escape System.

The system will use a single, fixed thrust (21,900 pounds), gimbaled engine with a specific impulse of 318.7 second. The engine has a 750-second service life and a minimum capability of 50 restarts. The system uses earth-storable hypergolic propellants (50 : 50 mixture hydrazine and unsymmetrical dimethylhydrazine fuel and nitrogen tetroxide oxidizer); a pressurized propellant feed system, with a nozzle expansion ratio of 60:1; and an ablation-cooled 100-psia thrust chamber.

Contract for the Service Module Propulsion System rocket engine was awarded to Aerojet-General Corporation in April 1962.

The pressurization and propellant supply subsystems were studied and a decision was made to proceed with a design using multiple tanks with a series transfer and feed subsystem. The tank configuration and arrangement was established using four tanks of equal length, fuel tanks diametrically opposed and oxidizer tanks diametrically opposed. Tank capacity was increased from an initial capacity of 39,500 pounds to 45,000 pounds of usable propellant.

Full scale injector firings using modified Titan engine steel blanks with workhorse steel chambers were made with four different injector orifice patterns. Tests on three patterns were characterized by unstable operation while stable operation was obtained with the fourth, a long impingement triplet pattern. Full-scale aluminum developmental injectors are being fabricated and are slated for testing in early October 1962. Ablative thrust chamber tests with 2,200-pound-thrust engines have demonstrated the ability of prototype ablative materials to withstand 750 seconds of operation at sea level conditions. Simulated altitude testing of similar small scale ablative chambers will be accomplished at the USAF Arnold Engineering Development Center during November and December 1962. Purchase orders were released to Lear Siegler, Inc. for pre-prototype gimbal actuators and amplifiers. The detail design of an engine simulator for testing the thrust mount and gimbal actuation subsystem is 50 percent completed.

Manufacturing commenced on three sets of battleship-type propellant tanks for use as propellant subsystem development test fixtures. Preliminary detail drawings of propellant system piping and components are being prepared; and laboratory tests are being conducted to determine material compatibility, fabrication processes, and cleaning and purging techniques.

Communication System

The Communication System for the Command Module transmits voice, telemetry, and television signals to the Ground Operational Support System (GOSS) stations from lunar distance; receives and demodulates voice transmission from the GOSS stations; and receives and re-transmits in phase-coherence, a pseudo-noise coded signal, to enable selected GOSS stations to track the spacecraft in angle, range, and range rate during deep space phases of the mission. The system provides voice communication between crew members both inside and outside the Command Module, the transfer of audio signals to and from transmitters and receivers, and radio direction-finding and voice for recovery phase.

North American Aviation has subcontracted the Command Module Communication System to Collins Radio Corporation. Exceptions to this are the DSIF high gain and C-band transponder antenna systems which are retained by North American Aviation and the personal communication equipment which is government furnished equipment (GFE).

The Communication System is comprised of thirteen principal components:

a. Pulse Code Modulator Telemetry Equipment - This equipment will prepare data, guidance, voice and television signals, both digital and analog, for transmission to earth over either the VHF or UHF telemetry links. Proposals for this equipment have been received and are presently being evaluated by Collins Radio Corporation.

b. Deep Space Instrument Facility Transponder and Power Amplifier - A Deep Space Instrument Facility (DSIF) transponder system will transmit data and television signals, range information, two-way voice, and provide signals for coherent doppler tracking during translunar and transearth trajectory phases of the mission. During earth-orbital missions, the transponder will be used for transmission of television signals. Proposals for this equipment have been received and are presently being evaluated by the Collins Radio Corporation.

c. C-Band Transponder - The C-band transponder will enable radar tracking during near-earth phases of flight and will be compatible with existing GOSS equipments. Proposals for this equipment have been received and are presently being evaluated by Collins Radio Corporation.

d. VHF Recovery Beacon - The VHF beacon will provide aid to rescue aircraft in locating the Command Module during the post-landing phase. North American Aviation and Collins Radio Corporation have been given requirements for the recovery beacon. Procurement specifications are being prepared.

e. Television Equipment - The television equipment will provide for the transforming of visual information to electrical signals suitable for transmission to earth. An analog TV system has been selected rather than a digital system because less development is required for the spacecraft system as well as the ground system. Closed circuit TV for the spacecraft system will not be provided. Collins Radio Corporation is preparing procurement specifications for this system.

f. Intercommunication Equipment - This equipment will enable the routing of voice data to and from the several voice transmitters and receivers on the spacecraft. Development of this equipment at Collins Radio Corporation is on schedule.

g. VHF/FM Transmitter - This equipment will be used to transmit telemetry and voice data to the GOSS network when the spacecraft is operating at near-earth distances. Development of this equipment at Collins Radio Corporation is on schedule.

h. VHF/AM Transceiver - This unit will enable two-way voice communications with the GOSS at near-earth distances and with the Lunar Excursion Module and crew outside the Command Module. Development of this equipment at Collins Radio Corporation is on schedule.

i. Personal Communication Equipment - This equipment will enable voice communications between the crewmen while external to the spacecraft, also between crewmen and the Command Module and/or the Lunar Excursion Module, and will provide for physiological data transmission to the Command Module and Lunar Excursion Module. A decision was made to provide full duplex voice with a separate channel for biomedical data. Based on this decision, design of the spacecraft portion of this equipment can proceed.

j. HF Transceiver - This equipment will enable radio direction finding and voice communications during the post-landing phase. North American Aviation has been directed to provide a 5-watt AM transceiver in lieu of the proposed 20-watt single sideband transceiver. This transceiver will be electrically equivalent to the recovery transceiver being used in Project GEMINI. Development is on schedule.

k. VHF/UHF Omnidirectional Antenna Equipment - This equipment will enable near-omnidirectional transmission and reception of VHF and UHF radio frequency energy. Developmental work on this antenna indicates that it will be available for early boilerplate spacecraft.

l. UHF Directional Antenna Equipment - This equipment will enable high-gain transmission and reception of UHF RF energy during translunar and transearth phase of the flight. Design requirements are being established by North American Aviation.

m. C-Band Antenna Equipment - This equipment will enable near-omnidirectional transmission and reception of the C-band tracking RF energy. Design requirements are being established by North American Aviation.

In order to conserve weight and space, no redundant communication subsystems are now planned for spacecraft incorporation. The items to be measured have been decreased by approximately 50 percent (518 to 260) from earlier estimates, thus allowing the telemetry bit rates to be decreased to 32,000 bits/second. This reduction in bit rates has resulted in a weight and power savings.

The MSC is considering the desirability of initiating a North American Aviation contractual effort to develop a unified system and eliminate the currently programed non-unified system for support of the APOLLO vehicle while it is in earth orbit. Such a system will allow the use of the lunar mission transponder for near-earth transmissions and eliminate the several transmitters which are required by the present concept. The elimination of transmitters will reduce the spacecraft weight approximately 100 pounds and will also reduce the complexity and cost of the spacecraft system.

Operational Instrumentation System

The Operational Instrumentation System (OIS) for the Command Module will convert those physical parameters to electrical signals which must be displayed, recorded, or transmitted. The conversion of the parameters will be accomplished by sensor equipment such as accelerometers, potentiometers, and strain gauges. The conversion will also require a combination of resistors, amplifiers, frequency converters, or demodulators for use as signal conditioning equipment which will condition all analog signals to a standard level and form before commutation in the telemetry equipment. This signal conditioning will allow convenient display, recording, and transmission of the information. The OIS will also provide stable frequencies for synchronization of time-dependent spacecraft subsystems, except for the guidance and navigation equipment, and provide a time reference for all time-dependent spacecraft operations through the use of spacecraft central timing equipment. The system also provides data-storage equipment to store, for future readout, data which cannot be transmitted to the ground in real-time, plus playback capability of analog and digital data.

Spacecraft central timing equipment is under procurement by the prime contractor, but subcontractor selection has not yet been made.

Data-storage equipment has been included in the Communications System contract awarded to Collins Radio Corporation; bids for the recorder have been received by Collins Radio Corporation and a contractor is being selected.

The remaining items for the OIS are relatively short lead-time items, and will be procured by North American Aviation, Inc. The overall system status is on schedule.

Electrical Power System

The Electrical Power System supplies, regulates, and distributes all electrical power requirements of the Command Module and Service Module for the full duration of the mission, including the post-landing period. From prelaunch until reentry, the average normal electrical load is 1630 watts, and the average emergency load is 1200 watts. During reentry and recovery the normal and emergency loads average 670 and 540 watts, respectively. Post-landing loads average 22 watts normal and 14 watts emergency.

The major components of the system include hydrogen-oxygen fuel cells, hydrogen-oxygen reactants and their storage, batteries, and space radiators.

The three hydrogen-oxygen Bacon fuel cell batteries supply the main and emergency power throughout the mission except for the earth reentry phase. Two of the fuel cells can carry normal electrical loads, and one will supply emergency power. The fuel cell contract was awarded to Pratt and Whitney in March 1962. Design and development has proceeded to the testing of a 31 cell "stack" and other system components. Performance predictions have been met and exceeded in single cell tests. Complete module tests will begin during the next quarter.

The hydrogen and oxygen reactants for the fuel cell power supply are stored in the supercritical state in spherical pressure vessels. The oxygen vessel also supplies gas to the Environmental Control System.

The supercritical storage system subcontract was awarded to Beech Aircraft, Boulder, Colorado. A recent decision was made to provide heat input to the storage vessels with electrical heaters rather than the water-glycol loop.

Three zinc-silver oxide batteries supply power for all electrical loads following separation of the Service Module and during the brief periods of peak load. One of these batteries is reserved exclusively for the post-landing loads. Eagle-Pitcher was selected in August 1962 as subcontractor for the batteries. Two battery chargers are provided to recharge the batteries in flight from the fuel-cell power supply. Three 1250VA, solid-state inverters provide 400 cps, 115/200 volt, three phase AC power by inversion of a portion of the fuel cell DC power source.

Space radiators, mounted on the Service Module exterior, reject to space the waste heat generated by the fuel cells. A water-glycol loop provides the heat-transfer medium. The water generated by the fuel cells is recovered and stored aboard the spacecraft for crew use and for supplemental evaporation cooling. Radiator design is underway. The radiator has two panels, totalling 34 square feet, and is designed to accommodate a maximum 30-psi pressure drop.

An incandescent light system has been selected over a fluorescent system because of its comparative simplicity, lesser volume requirement, and lighter weight.

All pyrotechnic functions will be initiated with hot-wire squibs, except for the launch escape motors which will use explosive bridge-wire firing units. Design has been completed on Boilerplate No. 6 (Pad Abort Test) sequencer, testing has been conducted on the solid-state sequencer for Boilerplate No. 13, and hardware has been delivered for the development of the initiation system for the Launch Escape System motors.

Earth Landing System

The Earth Landing System stabilizes the Command Module during the subsonic phase of the descent and reduces the velocity at landing to acceptable crew tolerances. The landing system consists of one 13-foot subsonic drogue parachute and three 88-foot ring-sail parachutes. It is housed in the apex of the Command Module and is readied for deployment by jettisoning the apex heat shield. The apex heat shield is normally jettisoned at about 40,000 feet, the drogue parachute is deployed at 25,000 feet, and the main parachutes at an altitude of 15,000 feet. For atmospheric aborts below 60,000 feet, the apex heat shield is thrust into latches on the launch escape tower which removes the heat shield during the tower jettison operation. Above 60,000 feet in the boost trajectory, the abort mode must be changed to retain the heat shield to prevent parachute heating. This change prevents drogue parachute deployment before turning the module around. The RCS is incapable of reorienting the Command Module below 100,000 feet because of a strong static trim point with the Command Module apex forward. Therefore, a period exists where abort cannot be accomplished. System modifications to eliminate this problem are being studied.

The change from the initial approach to the above system was adopted because removal of the apex heat shield by the drogue parachute in the original system presented a high probability of recontract with the main parachutes. Reliability was also enhanced by using individual mortar-deployed pilot parachutes to extract the main parachutes. The capability of deploying the main parachutes independently of the drogue parachute led to the elimination of the redundant drogue parachute.

The parachutes will be qualified for use with a 9,500-pound Command Module which provides a weight growth contingency. Parachute designs in two fabric weights are complete.

Eight single main parachute drop tests have been conducted. Parachute performance and reefing parameters have been successfully investigated at design q and 1.25 design q.

ADAPTER

The adapter is the structural attachment between the spacecraft and the launch vehicle. Two basic adapters are required, one for Saturn C-1 and one for Saturn C-5.

The C-1 adapter is a cylindrical bonded-aluminum honeycomb-sandwich structure. The sandwich core is 1.00 inch in depth with .018 inch thick 7075 Aluminum Alloy face-sheets bonded on the structure. Normal separation is accomplished by cutting continuous structure with flexible linear-shaped charges. The present staging sequence for a high altitude abort is to blow out panels in the adapter and separate from the booster with the Service Module Propulsion System. Because ignition effects on the nozzle are difficult to predict, and alternate method is being investigated in which posigrade rockets will be used to achieve some separation from the launch vehicle before the Service Module Propulsion System engine is started.

The C-5 adapter will enclose and support the Lunar Excursion Module, and, therefore, is in the early predesign stages. It will be conical in shape, utilizing bonded-aluminum honeycomb-sandwich construction. It may be desirable to split the adapter longitudinally for repositioning the Lunar Excursion Module.

LUNAR EXCURSION MODULE

The Lunar Excursion Module (figure 4) serves as a shuttle vehicle for transferring two of the three crew members and payload from the Command Module in lunar orbit to the lunar surface and their return to the Command Module. Included in this operation are the functions of separation from the Command Module, lunar descent, lunar landing, ascent, and rendezvous and docking with the Command Module.

General design criteria in the form of technical guidelines for the Lunar Excursion Module and system performance requirements have been established and incorporated in a Statement of Work (ref. 1). A request for proposal was submitted to industry in July. Proposals were received from nine industrial firms in early September. Evaluation of the proposals has been completed and reported to the NASA Administrator.

The Lunar Excursion Module consists of twelve principal systems: Guidance and Navigation, Stabilization and Control, Propulsion, Reaction Control, Lunar Touchdown, Structure including landing and docking system, Crew, Environmental Control, Electrical Power, Communication, Instrumentation, and Research and Development or Scientific Instrumentation. The systems requirements are presented in the Lunar Excursion Module Statement of Work and will not be reproduced here. Pertinent additional comments on the various systems are presented in the following paragraphs. A consideration of prime importance to practically all systems is the possibility of using components now existing or under development for Project MERCURY, Project GEMINI or the Command and Service Modules. A study of these possibilities is nearly complete.

Structural System

In addition to the fundamental load carrying structures, the Lunar Excursion Module structural system includes meteoroid protection, radiation protection inherent in the structures, and passive heat protection.

Lunar Touchdown System

The Lunar Touchdown System arrests lunar impact, supports the Lunar Excursion Module during its period on the moon, and provides a lunar launching base.

The touchdown system, as a modular part of the Lunar Excursion Module, is stowed in the adapter prior to activating the Lunar Excursion Module. Manual deployment of the Lunar Touchdown System is under consideration. Fixed landing gear legs are a possibility depending upon the overall size of the Lunar Excursion Module.

The Lunar Touchdown System may use a crushable material in the landing legs.

Crew Equipment

Crew Equipment for the Lunar Excursion Module is comprised principally of the same type functional equipment as for the Command Module. In addition, large windows in the Lunar Excursion Module will permit the two-man crew to exercise a high degree of direct visual control of the lunar touchdown.

Environmental Control System

The Environmental Control System (ECS) for the Lunar Excursion Module provides an "open face plate" environment in the manned compartment. The system has the same characteristics as the Command Module ECS. A self-contained portable life-support system (backpack) provides life support for the crewmen during lunar exploration outside the vehicle; the portable life-support system provides 4 hours operation "per charge," and is rechargeable from Lunar Excursion Module ECS supplies.

Guidance and Navigation System

It is expected that the Lunar Excursion Module guidance system will use as many of the components as possible which are identical to those in the Command and Service Modules. Studies at the Massachusetts Institute of Technology (MIT) have indicated that those changes required will be towards simplification of the computer while using the presently conceived inertial measurement unit and the scanning telescope. There is no requirement for a sextant onboard the Lunar Excursion Module.

The radar required for use on the Lunar Excursion Module is presently a part of the Lunar Excursion Module contractor's responsibility. After the contractor is selected, types of radar to be used will be determined on the basis of MIT prepared specifications.

Stabilization and Control System

This system provides the necessary stabilization and control for the Lunar Excursion Module during all phases of its operation. These phases of operation are the lunar transfer orbit to the moon, lunar landing, lunar launch, and lunar transfer orbit to the Command and Service Module which includes rendezvous and docking. It is expected that the Stabilization and Control System of the Lunar Excursion Module will use as many of the identical Command and Service Module components as possible.

Reaction Control System

The Reaction Control System (RCS) provides thrust impulses for attitude control and stabilization during the active phases of the Lunar Excursion Module and minor translational capability for terminal rendezvous and docking of the Lunar Excursion Module with the Command and Service Modules. The system uses hypergolic bipropellants and is pulse modulated. The system is located on the launch stage and has a separate propellant storage and pressurization system. Emergency feed from the launch stage propellant tanks is also available.

Propulsion System

The Propulsion System of the Lunar Excursion Module provides thrust requirements for lunar landing, lunar launch, and lunar orbit rendezvous with the Command and Service Modules. The system uses earth-storable hypergolic propellants and has a pressurized propellant feed system. The present approach involves a fully-staged propulsion system for the descent phase of the mission. The descent engines will require the development of an engine capable of being throttled to an 8 to 1 ratio. Different approaches for achieving this capability are under considerable study. A fixed thrust engine will be used for ascent propulsion.

Communications System

The Communications System for the Lunar Excursion Module will transmit voice, telemetry, and television signals to selected GOSS stations; receive and retransmit in phase coherence, a pseudo-noise coded signal, which enables selected GOSS stations to track the spacecraft in angle, range and range-rate during mission deep space phases. The system will also enable voice communication between crew members both inside and outside the Lunar Excursion Module and between the Lunar Excursion Module and the Command Module when the Command Module is within line of sight.

Operational Instrumentation System

The Operational Instrumentation System for the Lunar Excursion Module will convert physical parameters to electrical signals for display or transmission and, also, obtain photographic records of events occurring inside and outside the Lunar Excursion Module. The system will also condition electrical signals to a common level for convenient display and transmission, and will provide stable frequencies for sychronization of time-dependent Lunar Excursion Module subsystems except for guidance and navigation equipment. In addition, it will provide a time reference for all time-dependent Lunar Excursion Module operations.

Electrical Power System

The Electrical Power System supplies, regulates, and distributes all electrical power required by the Lunar Excursion Module for the duration of its specific mission. Preliminary studies indicate that electrical loads will be less than 2 kilowatts during flight phases and approximately 800 watts during the lunar stay. Candidate power sources include open-cycle and closed-cycle fuel cells or chemical dynamic engines for the long-duration electrical power needs and silver-zinc primary batteries for lunar launch and rendezvous.

SPACE SUIT SYSTEM

The APOLLO Space Suit System provides the crew with an extravehicular capability to accomplish short exploratory examinations of the lunar surface, and, as a secondary objective, allows long term emergency protection in the event of cabin depressurization. The suit also provides for extra-vehicular operations to accomplish spacecraft maintenance in free space.

The Space Suit System consists of a full pressure (pneumatic), closed circuit, anthropomorphic type garment and its components; a constant wear garment; a coverall to provide additional environmental protection, and a portable, self-contained life support system. The system will weigh approximately 60 pounds but will allow unassisted ingress and egress to the spacecraft.

Since the cabin environment is normally a "shirt-sleeve" condition, the space suit is designed for rapid donning: 5 minutes, unassisted. The design of the suit will take into consideration an operating pressure of 3.5 psia (4 hour comfort; 4 day tolerable) with 100-percent oxygen pressurizing gas; waste and hygiene requirements; operational requirements in free space and lunar temperature range, thermal control, mobility, noise and buffet protection requirements; communication and telemetry; emergency pressure protection; vision; feeding; and waste removal.

The portable life support system will provide respiration and ventilation requirements along with CO_2, H_2O, and body contaminate removal, and thermal control for a minimum of 4 hours continuous operation without recharging; the system will be designed to function in any plane or position, in zero gravity and high acceleration environments.

Hamilton Standard was selected as the prime contractor for the space suit assembly in July 1962. The principal subcontractor is International Latex Corporation, who will fabricate the pressure garment.

A detailed design study and an analysis of various suit components proposed for the initial prototype suit and portable life support system are underway.

Study results are to be completed by the contractor by November. Four prototype suits and two prototype support systems will be furnished by the contractor by June 1963 for the MSC static testing and evaluation.

Procurement is underway for interim pressure suits to support APOLLO test programs.

R&D INSTRUMENTATION/COMMUNICATION SYSTEM

The basic Research and Development Instrumentation/Communication System will provide a reliable, fully qualified data measuring, processing, storing and transmitting subsystem, and a tracking subsystem. Using modular construction, the tracking subsystem will have sufficient inherent flexibility to satisfy any of the varying mission requirements of the several APOLLO R&D and qualification flights. The basic system, however, will be tailored to meet the specific objectives of each flight.

Those physical parameters which must be recorded and/or transmitted will be converted to electrical signals through the use of sensor equipment such as accelerometers, strain gages, thermocouples, and so forth. Signal conditioning equipment consisting of combinations of resistors, capacitors, amplifiers, frequency converters, and so forth will be used for conditioning signals to a standard level and form before commutation in the telemetry equipment for convenient recording and/or transmission. A stable reference frequency for use in correlating data recorded on board the spacecraft with that recorded by ground data acquisition will be provided by a coded time generator. Data storage equipment will store for future readout, analog data which cannot be transmitted to the ground in real time or which is being stored for backup purposes. Two C-band radar transponders and associated antenna subsystem components will enable the FPS-16 ground radar to track the spacecraft to far greater ranges than is possible by skin-tracking alone. Command receiver/decoders are provided on those flights in which a radio command capability is required for ground-to-spacecraft command. The telemetry system will employ standard IRIG FM/FM modulation techniques in conjunction with 90x10 and 90x1$\frac{1}{4}$ commutators providing pulse amplitude and pulse duration wave trains.

The R&D instrumentation/communication systems were designed and developed by the MSC and will be GFE to North American Aircraft. Evaluation and selection of the system components have been completed

All procurement for the first six flight spacecraft has been completed. Initial delivery of fully qualified flight hardware from the MSC to North American Aviation for the first boilerplate R&D spacecraft has begun.

SCIENTIFIC INSTRUMENTATION

Preliminary investigation into types of scientific experiments which will be performed on lunar flights has been conducted. When firm requirements are established, hardware implementation will begin.

Maximum volume, weight, and power allotments of 10 cu. ft., 250 pounds, and 2400 watt hours are currently provided in the Command Module and the Lunar Excursion Module for the inclusion of the scientific instrumentation when required.

Allocations were based on the following anticipated types of scientific instrumentation: cameras, GM counters, thermocouples with probes and recorders, geological pick and net container, signal-strength meter, microscopes, drag bars, soil samples, explosive charge drill and core sample devices, seismographs, mass spectrometers, "red-head" gauges, gravimeter or pendulum system, and magnetometer.

All scientific instrumentation equipment will be government furnished equipment.

WEIGHT

The APOLLO spacecraft weights are shown below. A weight apportionment has been made based on an assumed 90,000-pound payload performance capability of the Saturn C-5. This weight is termed a design allowable. A lower target weight for each module has been assigned. Achievement of the target weight will allow for increased fuel loading and therefore greater operational flexibility and mission reliability. The Service Module propellant tanks have been sized to take advantage of achieving target weights or increased C-5 performance over 90,000 pounds. The current spacecraft weight falls between the design allowable and target weights.

	Design Allowable	Target
Command Module	9,500	8,500
Service Module (includes trapped or unused propellant)	11,500	11,000
Adapter to S-IVB	3,200	3,200
Total	24,200	22,700
Service Module Useful propellant	40,300	37,120
Total	64,700	59,820
Lunar Excursion Module	25,500	24,500
Total	90,000	84,320

Weights presented above are largely calculated, estimated, or taken from specifications to contractors. Little hardware has been actually weighed to this date. A weight-control program has been implemented and is favorably influencing the weight-time history.

FLIGHT TECHNOLOGY

Mission Natural Environment

The original definition of the external natural environment of the spacecraft was contained in the December 18, 1961, Statement of Work. These environments were used in the early APOLLO design work with the result that the micrometeoroid, solar proton radiation, and lunar-surface characteristics were found to be most critical to the spacecraft design.

The micrometeoroid environment specified in the original Statement of Work was the Whipple distribution with a mass of 25 grams for a zero magnitude meteoroid. This environment was modified as the result of later data and analysis to 2.5 grams for a zero magnitude meteoroid and appeared in the Lunar Excursion Module Statement of Work. The density of the meteoroids was assumed in each case to be that of stone: 3.5 grams/ccm.

There is some evidence that indicates that the environment in the Lunar Excursion Module Statement of Work may still be too severe. At the present time, efforts are being made to accumulate sufficient information to substantiate a further reduction in the severity of the micrometeroid environment.

There is also a large uncertainty in the estimates of the penetration caused by a particle of given mass, density, and velocity. Tests in ballistic ranges have been carried out at Ames Research Center, Langley Research Center, and North American Aviation. However, these facilities can achieve only about half of the expected velocity of 75,000 feet per second for the critical size meteoroids. A major improvement in the capability to predict penetration will come from a test program presently being negotiated by North American Aviation with General Motors Corporation. The General Motors Corporation facility is capable of achieving particle velocities of 75,000 feet per second.

The original Statement of Work for both APOLLO and the Lunar Excursion Module did not contain any quantitative requirement for achieving a given probability of the spacecraft not being punctured by meteoroids. Without a requirement, it is impossible to determine whether the present structural design is adequate from this standpoint. At the present time, estimates are being prepared which will show the implications of specifying a given probability of no puncture as a function of the various proposed meteoroid environments and penetration equations.

The solar-proton-flux component of the space environment was specified in the original Statement of Work in terms of a "Typical solar event."

Calculations based on this event indicate that additional shielding will be required to maintain allowable dose levels for this event. However, recent data have indicated that the proton-flux-level of the event in the Statement of Work is too high and that only very little, if any, selected body shielding will be required. It is anticipated that the Statement of Work description of solar-proton-flux environment will be revised in the very near future.

The recent appearance of an artificial, trapped-radiation belt around the earth has required a re-evaluation of the radiation doses encountered while the regions of trapped radiation are being traversed. This evaluation is presently underway and is to be completed within a month.

The first issue of a document defining the requirements for Project APOLLO and, also, a further definition of the natural environment was issued in August 1962. This document summarized the areas in which additional definition would be useful for Project APOLLO design. No areas were found where additional information on environments would be a prerequisite for attempting the Project APOLLO lunar landing mission.

Performance

The analysis of the launch escape configuration performance using available data has been virtually completed. The Launch Escape System, employing a kicker rocket for obtaining increased initial angular rates, is capable of achieving an apogee of 4,000 feet at a lateral displacement of 3,000 feet for pad aborts. Performance of the Launch Escape System at maximum q will provide a separation between the Command Module and Service Module booster of 90 feet in the first second and 195 feet at the end of 1.5 seconds.

The Service Module Propulsion System requirements were established by the NASA for the lunar orbit rendezvous mode. A total of 3,883 feet per second was specified for maneuvers with the Lunar Excursion Module attached to the Command Module and 4,801 feet per second for maneuvers without the Lunar Excursion Module attached. For a 90,000-pound injected payload gross weight, 40,300 pounds of fuel will be required by the Service Module Propulsion System. To provide room for growth and payload weight increases, the Service Module tanks have been sized to carry 45,000 pounds. The Service Module RCS ΔV and $\Delta\dot{\theta}$ requirements have also been estimated. The resulting RCS fuel required is 965 pounds.

The Lunar Excursion Module performance characteristics as specified in the Lunar Excursion Module Statement of Work, include a propulsion system capability of 7,740 feet per second prior to lunar landing and 6,880 feet per second after lunar take-off.

The reentry performance requirements of the Command Module presently include a 5,000-nautical mile reentry range capability. For the design L/D of 0.5, the 10 G corridor width is 30 nautical miles, and the lateral maneuverability is ± 400 nautical miles for entry at escape velocity. The trimmed L/D is directly dependent on the vertical center of gravity offset. Because of difficulties encountered in attaining this center of gravity offset, present studies are being directed at defining the minimum allowable L/D of the reentry vehicle.

Spacecraft Dynamics

The principal areas of investigation to date in the area of spacecraft dynamics have been Launch Escape System (LES) aborts, "turnaround" after aborts, and spacecraft separation from the final booster stage.

Estimates of LES abort dynamics characteristics are complicated by unknown effects of the LES rocket jet during rocket burning. Until jet-on wind-tunnel tests are completed, the jet-on characteristics must be obtained analytically. With the estimated jet effects, the escape configuration can permit $\frac{1}{2}^\circ$ LES thrust vector misalignment in the center of gravity and still achieve adequate abort performance.

For LES aborts at altitudes above 130,000 feet, the dynamic pressure becomes low enough to permit the expected LES thrust misalinements to cause tumbling with the resulting loss of attitude reference. Both aerodynamic and pilot-controlled means are being investigated for recovery of proper reentry attitude reference.

The "turnaround" maneuver is required after non-tumbling aborts to orient the Command Module with blunt face forward and thereby permit removal of the apex cover and deployment of the main parachutes. Because of the existence of an apex-forward trim point, the Command Module cannot be reoriented by the RCS at dynamic pressures greater than 20 pounds per square foot. Both aerodynamic fixes and recovery sequence modifications are being investigated to solve this problem.

Several separation techniques for the separation of the spacecraft from the booster at abort and at normal mission injection have been investigated. Some of the techniques considered were posigrade solid rockets on the Service Module adapter, retrograde rockets on the S-IV and S-IVB, and the use of Service Module fire-in-the-hole retrorockets. Posigrade rockets on the adapter appear to be the most desirable system because of the weight advantage over the retrograde system and the long development time of the fire-in-the-hole technique.

Aerodynamics

The aerodynamic characteristics of the various APOLLO Spacecraft configurations have been determined in the first phase of the wind-tunnel program. The Command Module has an L/D of 0.5 at an angle of attack of 33°. The center of gravity must be offset 9.9 inches from the geometric centerline at station 1040 to achieve trim at this agnle. The design reentry W/C_DA when L/D=0.5 is 75 pounds per square foot.

The launch escape configuration aerodynamic characteristics show the characteristic instability caused by the addition of the escape rocket and tower. Wind-tunnel tests of flow separators at the base of the LES rocket show that static aerodynamic characteristics are greatly improved by these devices. A flat disc of 3.0 rocket case diameters improves static stability to the same degree as about 400 pounds of ballast. Acceptance of this aerodynamic fix will depend on the results of forthcoming wind-tunnel tests to determine the flow separator effects on dynamic stability, noise and buffet during launch, and launch-vehicle control characteristics.

The potential problems of aeroelastic coupling of the C-1 Saturn-APOLLO configuration are to be investigated with an 8-percent elastic model in the Langley Research Center 16-foot wind tunnel. This model has been fabricated and is presently being checked out at North American Aviation. Rigid mode as well as first and second structural bending modes are simulated. Test results are expected by the end of November.

The Project APOLLO wind-tunnel program is presently in its eighth month. To date, 2,800 hours of time have been used in approximately 30 facilities, both government and private. There are 18 force models, 16 pressure models, and 8 heat-transfer models. Aerodynamics of the launch escape configuration are, in general, defined throughout the angle-of-attack range, $0° < \alpha < 90°$, and over the Mach number range, $0 < M < 9$. Command Module aerodynamics are defined for Mach numbers up to 9 over most of the angle-of-attack range, $0° < \alpha < 180°$, while definition of limited angle-of-attack ranges at high Mach numbers has been accomplished. Blanket plots of both Command Module and LES aerodynamics are presently being prepared. Future plans call for updating all models to final configuration design as soon as exterior design freezes are accomplished.

Aerodynamic Heat Transfer

The reentry heating conditions have been estimated for several reentry angles, reentry ranges, and reentry guidance modes. The most critical heat-shield design heating condition is the maximum range (5,000 nautical miles) reentry from the overshoot entry angle ($\gamma = -5.5°$). For this condition the total stagnation-point heat load is approximately 130,000 BTU/ft^2 and the total heat load to the entire vehicle is 5.5 million BTU.

Convective heating distributions around the vehicle indicate that the windward-meridian-afterbody heating rates are 30 percent or less of the stagnation-point heating rate, and the leeward-meridian-afterbody heating rates are 5 percent or less.

The non-equilibrium radiation heating was once thought to be a severe problem for blunt vehicles of the APOLLO type. Present estimates based on recent research results at AVCO indicate the most critical non-equilibrium heating rates which occur on overshoot trajectories are less than the equilibrium radiation heating rates of about 200 BTU/ft^2/sec on the 10g undershoot trajectory.

Exit and abort conditions have been calculated and do not represent critical heating conditions for the Command Module. Heating of the LES during launch and aborts will be sufficiently high to require selective shielding on the tower structure.

Heat-transfer tests have been completed for the Command Module in the Mach number range of 6 to 10 and over the angle-of-attack range, $140° \leq \alpha \leq 180°$. Limited tests have also been performed at Ames Research Center in the Mach number range near 14. In the next few months, tests will be completed in the hypersonic Mach number range from M=11 to M=20, and also the supersonic Mach number range, M=1.5 to M=5.0.

Additional testing is being done at AVCO in the 1.5-inch and 6.5-inch shock tubes.

Ablation Material Thermal Performance

The charring ablative material presently being used for reentry heat protection of the Command Module is AVCO 5026. This is an eposy resin impregnated with silica fibers and phenolic microballoons. The epoxy resin provides large quantities of gas while decomposing which, when reaching the boundary layer, block a portion of the heat input. The silica fibers are added to improve the mechanical strength of the char layer. The presence of a char layer results in high surface temperatures and therefore reradiation of a substantial portion of the heat input. The addition of the phenolic microballoons provides a controlled means of introducing air spaces in the material thereby reducing material density, which at present is 55 lb/cu ft. Recent modifications to the blending process of the materials (dry blend) indicate that densities of about 35 lb/cu ft may be achieved without significantly changing thermal performance.

The experimental programs at AVCO for determining thermal and mechanical properties of the ablative material are nearly completed on the high density formulation of 5026 and are just beginning on the low density formulation. The North American Aviation portion will be confined to obtaining early design data and to limited monitoring of the AVCO results.

Several undesirable fabrication characteristics have resulted from the basic nature of the 5026 material. This has prompted a NASA sponsored materials-screening program with the objective of determining if other ablation heatshield materials, with more favorable fabrication characteristics, can be found which do not sacrifice thermal or mechanical performance in comparison to 5026. This screening program includes both thermal performance tests and micrometeoroid-impact simulation tests. Three materials have been chosen, two are elastomer-filled honeycomb and the third is a sprayable subliming material. This screening program will be completed around the end of October. Several additional materials, however, may be added to the present program.

RELIABILITY

The target reliability goal for design purposes stated in the Spacecraft Statement of Work for the APOLLO mission is 0.90. The probability that the crew will not have been subjected to conditions in excess of nominal limits is 0.90, and the probability that the crew will not be subjected to emergency limits is 0.999.

The initial work statement apportionment for the Lunar Excursion Module is 0.984 for mission success and 0.9995 for crew safety. Other major system elements will require reapportionment to reflect the lunar orbit mission.

The following activities have been completed:

<u>Command and Service Module Development - North American Aviation</u>

a. An initial reliability apportionment to the component level for most systems has been completed.

b. A reliability program and test plan has been completed.

c. Procedures and schedules for qualification and reliability tests have been initiated; the initial qualification status report has been completed showing components to be tested, testing environment and testing schedule.

d. Subcontractor reliability and test plans are under review by the principal contractor.

Navigation and Guidance System Development - Massachusetts Institute of Technology

a. An initial reliability apportionment to the major subsystem (computer, etc.) level has been completed.

b. The completion of a reliability program and test plans is awaiting increased definition and integration of plans with industrial support contractors.

c. An analysis of the navigation and guidance system apportioned reliability design goal indicates that inflight maintenance and replacement will be required to achieve the goal.

General Dynamics/Convair - Little Joe II

a. Reliability apportionments to the various parts of the Little Joe II launch vehicle are completed.

b. A reliability program and test plan have been completed.

SPACECRAFT-LAUNCH VEHICLE INTEGRATION

Spacecraft-launch vehicle integration effects the integration of the spacecraft to the launch vehicle and its associated launch and flight control ground support equipment. This activity is accomplished through five Manned Spacecraft Center-Marshall Space Flight Center (MSC-MSFC) coordination panels which have met on the average of three times. The panels are comprised of members from MSC, MSFC, and Launch Operations Center (LOC). The panels, their areas of responsibility, and actions to date are as follows:

a. Flight Mechanics, Dynamics, and Control Panel. - The panel's responsibilities cover the areas of interface and integration between spacecraft and launch vehicle for aerodynamics; guidance and control interfaces; wind-tunnel test programs of configurations having effects on both spacecraft and launch vehicle; and range safety. This panel has established and implemented a MSC-MSFC signed agreement on the ground rules, the areas of responsibility, and the project management to be employed in the spacecraft and launch-vehicle wind-tunnel test program. A major problem that this panel has identified and solved is one related to escape tower flexure and effect on angle-of-attack sensor, which is directly connected to the SA-5 control system. A second problem recognized by this panel is the jettisoning sequence of the spacecraft Launch Escape System. During this sequence a structural frequency occurs and introduces a complexity into the control system. Preliminary analysis, however, indicates that this problem is solvable. Another problem that is presently confronting this panel concerns the pressure and temperature fields encountered during an abort from an exploding booster. An investigation is underway to determine these effects. A recent problem assigned to this panel for solution is the separation method of S-IV and S-IVB stages from the payload.

b. Launch Operations Panel. - The panel is responsible for assuring that the spacecraft and launch vehicle launch ground support equipment are compatible. This responsibility includes establishing space and facility requirements for checkout and mating of the Space Vehicle; integrating the Space Vehicle operation and countdown plan; resolving tracking and data requirements during launch; and establishing ground safety and pad operational plans. A particular problem this panel has defined and documented is the location of the center access hatch of the Command Module in relation to the launch complex. Also, the panel has been confronted with the problem of space allocation in the blockhouse. To alleviate the space problem, the space and equipment requirements in the Launch Control Center for Vertical Launch Facility (VLF) 34 and 37 were defined and documented. Two further problems now defined and documented are the APOLLO fluid service requirements and present spacecraft power and cooling requirements on the launch pad. In addition, this panel is involved in problems related to cable routings and communication between various facilities. As a ground safety requirement, the panel has agreed that a total destruct system document for SA-5 through SA-10 would be prepared before the mission for the Program Requirements Document.

c. Electrical Systems Panel. - Panel responsibilities are to define electrical interfaces between the spacecraft, launch vehicle and launch ground support equipment; to analyze and implement requirements for electrical-systems design compatibility for systems checkout during mating of the spacecraft and launch vehicle; and to enforce range safety requirements connected with the electrical systems. Wiring requirements for the "Q-Ball" spacecraft interface have been established by this panel. The spacecraft-launch vehicle electrical interface problem has been resolved by using three 61-pin lanyard-operated Bendix connectors. Space requirements for the APOLLO spacecraft electrical support equipment on the VLF-39 launcher/transporter have been defined and documented. Another problem that has been solved is that of the power requirements for SA-5 electrical support equipment. This particular panel established that an

"S-IV burning" signal would be supplied to the APOLLO Command Module 5 to 10 seconds after ignition. The problem of furnishing cables between electrical checkout equipment and facility equipment was resolved. Uniform Space Vehicle ground requirements have been established and studies are underway to set compatible launch complex grounding requirements. An immediate problem now being solved concerns the power requirements for the APOLLO Command Module while in transit between launch facilities. Another problem under consideration is the electrical installation in VLF-37 umbilical tower.

d. Instrumentation and Communication Panel.- This panel's responsibilities are to define spacecraft-launch vehicle communication and instrumentation interfaces, to define and resolve frequency interface problems, and to establish an overall Space Vehicle radio frequency interference program. This panel has established the Space Vehicle command frequency. A problem relating to C-Band beacon requirements for SA-5 and SA-6 has been resolved by this panel. A requirement has been set that transmission of "Q-Ball" data will be via spacecraft telemetry on SA-5 and 8. The purpose of this arrangement is to obtain data after spacecraft-launch vehicle separation.

e. Mechanical Integration Panel.- This panel's responsibilities are to define mechanical and structural interfaces between spacecraft and launch vehicle and GSE; recommend safe handling procedures for fuels, explosives and similar materials as they affect the overall space vehicle; resolve adapter mounting and mating problems; and resolve mechanical and structural problems imposed on the Space Vehicle by abort conditions. This panel has resolved that vehicles SA-5 and 6 will have an air conditioning barrier diaphragm between the launch vehicle instrument unit and the spacecraft adapter. Saturn-APOLLO mechanical interface problems have been resolved between the launch vehicle and the spacecraft and a control drawing, which has been agreed on by MSC and MSFC, is being prepared. Fluid service locations on the spacecraft have been defined and it was noted that manual operating mechanical connectors were being planned for all spacecraft pneumatic and fluid servicing connections. This panel is in the process of defining environmental control requirements.

MANUFACTURING

Manufacturing and delivery schedules for ground and flight test boilerplates and prototype spacecraft are being developed. The following paragraphs give the status of current manufacturing efforts.

MOCK-UPS

A wooden mock-up of the interior arrangement of the Command Module was completed and delivered to MSC in early September. An identical mock-up was retained at North American Aviation for design control.

Seven additional Command Module and Service Module mock-ups are planned: Partial Service Module and partial adapter-interface, Command Module for exterior cabin equipment, complete Service Module, spacecraft for handling and transportation (two), crew support system, and a complete spacecraft.

A Mock-up of the Navigation and Guidance Equipment has been completed.

A wooden mock-up of the Lunar Excursion Module exterior configuration was fabricated by North American Aviation as part of an early study of spacecraft compatability requirements.

BOILERPLATES

Ground Tests

The first boilerplate Command Module, (No. 25) for water recovery and handling tests, was delivered to the MSC in mid-August. Flotation, water stability, and towing tests were conducted with good results.

The second boilerplate Command Module, (No. 1) for water and land impact tests contained earth-impact attenuation and crew-shock-absorption systems. It was drop tested on September 25 with good results. The structure experienced over 20 g's and the crew-couch simulator experienced between 5 and 10 g's. The crushable shock struts on the simulated couch yielded 2 inches at the pilot's head and 4 inches at the pilot's feet. The impact velocity was 23 feet per second at 20° cant angle. Additional tests are planned.

Manufacturing of the remaining boilerplate Command Modules for ground test purposes is progressing and no difficulties are anticipated in meeting the current schedule.

Flight Tests

Manufacturing of the pad abort boilerplate Command Module, (No. 6) for early qualification of the Launch Escape System will begin on October 1 and will be completed and prepared for delivery to White Sands Missile Range by Mid-April 1963. A pad-abort test of boilerplate Command Module, (No. 6) is schedules for May 15, 1963.

PROTOTYPE SPACECRAFT

Manufacturing of the first Command and Service Modules with partial systems for ground test is scheduled for completion in January 1964. Manufacturing of the first Command and Service Modules with complete systems for flight test is scheduled for completion in March 1964.

The manufacture of the first Lunar Excursion Module for ground test of complete systems is scheduled to be completed in March 1965, and the first Lunar Excursion Module for flight test is scheduled to be completed in October 1965.

QUALITY ASSURANCE

The APOLLO Quality Assurance Programs have been prepared on the basis of NPC 200-2, "Quality Program Provisions for Space System Contractors," which specifies the requirement for a quality control plan. Such plans for the Command and Service Modules and for the Little Joe II launch vehicle have been prepared by the respective contractors and the MSC has approved these plans. The Massachusetts Institute of Technology quality control plan has not been submitted.

Arrangements have been made with Western Operations Office (WOO) to handle Government inspection at North American Aviation/Space and Information Systems Division. The inspection plan format has been provided to WOO.

Delegations of the APOLLO surveillance functions have been made to the Department of Defense at the North American Aviation major subcontractor plants, the Massachusetts Institute of Technology industrial support contractors plants, and at the Little Joe II contractors plants. The NASA, however, has no formal agreement for the performance of field service functions with the Air Force, which is the cognizant agency at a number of major subcontractors. It is understood that such an agreement is being developed, although it is expected that present staffing of these groups may result in problems in performing the necessary inspection for the NASA.

Specific Activities include:

a. The material review program of the Space and Information Systems Division was reviewed and revised to be compatible with APOLLO and S-II requirements.

b. An inspection and acceptance of boilerplate Command Module No. 1 was held. The boilerplate will remain at the Downey facility of North American Aviation's Space and Information Systems Division for further tests.

c. The Space and Information Systems Division of North American Aviation has agreed to participate in an Interservice Data Exchange Program.

d. An inspection and acceptance of boilerplate Command Module No. 25 was conducted on August 8, 1962, after which the boilerplate was shipped to MSC.

e. A meeting was held at North American Aviation to familiarize Department of Defense personnel with the requirements of NPC 200-1.

CREW

REQUIREMENTS AND SELECTION

The project APOLLO manned missions will be piloted by the Project MERCURY and Project GEMINI Astronauts and additional astronauts as needed for later APOLLO flights. The present team of 16 astronauts will contribute in the design, development, and operational phases of APOLLO.

Requirements for the APOLLO three-man crew are similar to those for the MERCURY and GEMINI programs. In selection of participating crews, emphasis will be placed on the individual tasks assigned as Pilot, Co-Pilot and Systems Engineer as well as the ability for interchange of task.

The following requirements have been used in astronaut selection:

(a) Experience as a jet test pilot, preferably engaged in flying high-performance aircraft.

(b) Experimental flight test status through the military services, the aircraft industry, or NASA; or graduate of a military test-pilot school.

(c) Degree in a physical science, biological science, or engineering.

(d) United States Citizenship, less than 35 years of age at the time of selection, and height not over six feet.

(e) Recommendation by the applicant's present organization.

TRAINING

Project APOLLO training requirements will be influenced by the following factors:

a. Crew availability for training.- The current flight crew training program is primarily oriented toward the manned Project GEMINI flights, however the pilots are also contributing to the conceptual phase of Project APOLLO. As Project APOLLO flight hardware becomes available, the training program will be oriented toward the operational aspects of that project.

b. System operational complexity.- In general, Project APOLLO systems, as currently defined, are more complex that Projects MERCURY or GEMINI counterparts. In addition to the greater system complexity, there is a greater degree of crew control and participation in system function.

c. Inflight test and maintenance.- A number of systems have critical components whose mean time to failure will not allow the system to meet its reliability requirements without inflight maintainance. The crew's capacity to perform functional fault location and replacement operations require crew capability beyond flight operational skills and becomes a major element in mission success probabilities.

Training requirements planning is 40 percent complete. Preparation of specific materials will start the first quarter of 1964.

Presently approved training equipment includes part-task trainers and mission simulators. The earth launch and reentry, orbital and rendezvous, and navigation and trajectory control part-task trainers are fixed base, special purpose simulators. Their early delivery will allow intensive practive by the crew in those mission functions where crew activity is time critical or requires the development of particular skills. The mission simulators are fixed base with complete mission capability providing visual as well as instrument environments. Adequate simulation capability is available to provide intensive crew training in critical mission phases and full mission training for both flight crews and GOSS crews. A mission simulator will be located at the MSC in Houston, Texas, and at the Atlantic Missile Range (AMR).

The part-task trainers are under review in an effort to reduce the complexity and cost as proposed by the contractor and to determine whether to use the trainers provided by North American Aviation or sub-contractors. Design requirement studies are 85 percent complete on the trainers.

A request for Proposal will be released on the mission simulators in January 1963. Design requirement studies are 55 percent complete.

OPERATIONS

SPACECRAFT CHECKOUT

A plan for the use of integrated spacecraft checkout equipment, not unique to any particular systems, has been prepared by the MSC. The plan includes a provision for spacecraft checkout at the factory and the launch site using the same checkout equipment. This plan is now being reviewed by North American Aviation and the Massachusetts Institute of Technology for its effect on airborne equipment, and for recommendations as to the implementation of the spacecraft portions of the system.

PREFLIGHT

Preflight operations for Project APOLLO missions flown from Cape Canaveral will be a joint operation by the Launch Operations Center, Manned Spacecraft Center, and Marshall Space Flight Center.

Hangar S will be used for checkout of the first Project APOLLO boilerplate Command Module delivered to the Cape.

Launch Complex 34 will be used for early Project APOLLO missions. The particular spacecraft requirements on the complex are being determined.

A flow plan and facilities requirements for an operational spacecraft checkout at Cape Canaveral have been determined, and plans for implementation are underway.

Projects APOLLO and GEMINI requirements have been combined and will make common use of certain facilities such as the weight and balance facility.

FLIGHT

Flight operations of the Project APOLLO missions entail monitoring and directing the missions, and recovery of the crew and Command Module.

The supplementary equipment required to support the Flight Operations are an integrated mission control center, an integrated mission control center computer, and certain network remote-site stations being used for Project MERCURY and Project GEMINI missions.

In support of these requirements, a study, design, and implementation effort is underway.

A study contract with Philco WDL laboratories (NAS 9-366) was initiated to determine the APOLLO-GEMINI requirements in developing an integrated mission control center (IMCC) and also, data flow requirements. The basic requirements of the IMCC have been defined under this contract and a preliminary data flow plan has evolved. Work is progressing in both of these areas.

The Request for Proposal for the IMCC has been completed and reviewed by the MSC. Included with the responsibility for the development of a specific recovery concept are the development, implementation, and operation of the IMCC and recovery control centers. The Request for Proposal will be submitted to industry upon review by NASA Headquarters.

Contractors' proposals for the real-time computer for the IMCC have been evaluated and IBM has been selected for contract negotiations.

Lincoln Laboratories, under contract NAS 9-105, has started a study program to define on-site data processing requirements and in addition, a study of the problems associated with a unified tele-communications system. Such a system will permit the use of the lunar mission transponder during near earth operations and eliminate the general transmitters which are required by the present concept, thus reducing weight by approximately 100 pounds and also the complexity and cost of the spacecraft system.

The APOLLO network requirements have been defined on a preliminary basis. The formal requirements document is currently being compiled and is scheduled for completion in the near future. North American Aviation has developed plans which include the personnel, training, and scheduling required for a group of system monitors to support mission operations at remote networks.

FLIGHT TEST PROGRAM

A tentative Project APOLLO flight test schedule is shown in figure 5. It is expected that a definitive plan will be available by the next reporting period. Major features of the test program are described below:

a. Pad abort.- Two tests are planned which will simulate an abort on the pad. The purpose of these tests is to qualify the Launch Escape System and its associated sequencing.

b. Sub-orbital.- (Little Joe II Test Launch Vehicle.) Three sub-orbital tests are now planned with the objective of development and qualification of the Launch Escape System and qualification of the Command Module structure. Test conditions include maximum dynamic pressure for the Launch Escape System and Command Module structures testing, and high atmospheric altitudes for Launch Escape System testing. The latter test requirement is receiving further review.

c. Saturn C-1.- The present Project APOLLO requirements for the Saturn development flights are for the determination of launch exit environment on SA-6 with provisions for spare on SA-8. Flight test requirements on Launch Vehicles SA-7, SA-9, and SA-10 are to flight test components of and/or the complete Emergency Detection System. SA-10 will be used to check the Emergency Detection System closed loop, and abort would, in the event of launch vehicle malfunction, occur prior to launch escape tower jettison. An operational spacecraft configured for manned operation is scheduled coincident with Saturn SA-10 for December 1964. Three or four additional manned flights are planned at 3-month intervals. Testing with the C-1 will terminate with availability of the Saturn C-1B for manned flights which is now scheduled for May 1966.

d. Saturn C-1B.- The present flight schedule shows four launch vehicle development flights prior to the manned flight. Preliminary APOLLO flight test objectives for the unmanned flights are one launch environment with a spare and two launch vehicle Emergency Detection System flights. Preliminary flight test objectives for the manned flights are three Lunar Excursion Module qualification tests, starting in the second quarter of 1966, and an additional Command and Service Module test with two spares.

e. Saturn C-5.- Six Saturn C-5 launch vehicle development flights are planned prior to the manned flight. Preliminary APOLLO flight test objectives for the unmanned flights are two launch vehicle Emergency Detection System flights, one spacecraft launch environment flight, and three reentry qualification flights. Preliminary objectives of the manned flights are completion of Lunar Excursion Module qualification, lunar reconnaissance, and lunar exploration missions. Although the first manned flight is scheduled as the seventh C-5, a spacecraft suitable for manned flight will be available for use on the sixth C-5 to take advantage of possible earlier C-5 development.

LITTLE JOE II LAUNCH VEHICLE DEVELOPMENT

The Little Joe II test launch vehicle (fig. 6) will be used for developmental and qualification flight tests of the APOLLO spacecraft under high q and high altitude abort conditions at suborbital speeds. The Little Joe II is a fin-stabilized, solid-fuel launch vehicle similar in concept and about twice the size of the Project MERCURY Little Joe launch vehicle.

Three spacecraft flights are planned: High q aborts - August 1963 (Spare - October 1963), and June 1964; high altitude flight abort - September 1964. The rocket-motor combination for the high q abort flights will be one Algol and six Recruits, and seven Algols for the high altitude abort. The flight tests are to be performed at White Sands Missile Range, New Mexico. A Redstone launch complex at White Sands Missile Range is being modified for Little Joe II.

Letter Contract (NAS 9-492) for the airframe and launcher was awarded to General Dynamics/Convair in May 1962. Aerojet-General Corporation was awarded a Letter Contract (NAS 9-465) in April 1962 for the Algol ID rocket motors (first stage Scout) and the design and development of the canted nozzles. Contract definitization for both letter contracts is in progress and is scheduled for December 1962. Recruit rocket motors and nozzles will be purchased "off-the-shelf" from Thiokol Chemical Corporation.

The General Dynamics/Convair contract included a 30-day study to determine the control system requirements. No control system was required for the high q abort tests. A combined aerodynamic and SCOUT-type monopropellant system was selected for the high altitude abort flights. Design of the control system has been initiated.

Wind-tunnel tests to determine forces, moments and control effectiveness over a Mach number range from .06 to 4.6 have begun in the Langley Research Center wind-tunnel facilities.

Structural criteria and design have been completed and construction of the first vehicle has begun. The electrical system design has been completed and is under review by the MSC. The design of the launcher has also been completed and is to be submitted to MSC for review.

The launch vehicle instrumentation to be transmitted by the spacecraft telemetry system has been determined. Measurement requirements for the high q abort tests have been established as 8 continuous and 44 commutated at 10 samples per second.

Fabrication has begun on the first two Algol ID motors. Design of the canted nozzle has been completed and reviewed by the MSC.

Program Management Plan milestones and PERT networks have been established for the Little Joe II program.

FACILITIES

Industrial requirements at Downey, California, are manufacturing facilities, office space, data reduction equipment, integration and checkout facilities, space systems development facility, radiographic facility, plaster mock-up storage facility, and impact facility. The A & E design of most of the buildings is underway, installation of some of the bridge cranes in an existing manufacturing area has been completed, and the A & E design criteria for the space systems development facility are being formulated by North American Aviation.

A survey of possible industrial test areas for APOLLO propulsion development resulted in the selection of a section in the White Sands Missile Range for this facility. The selection was based to a great extent on safety requirements for the toxic propellants used by the APOLLO Spacecraft. Advance engineering has started for the development, of the site and the industrial facilities criteria are being defined.

The suborbital qualification flights using the Little Joe II launch vehicle will also be conducted at the White Sands Missile Range. White Sands Missile Range will be responsible for insuring that all modifications of the Redstone launch complex and support facilities are completed per NASA requirements.

The requirements of AVCO, Minneapolis-Honeywell, and Pratt and Whitney for machinery and equipment have been submitted by North American Aviation. Except for critical lead-time requirements, approval of these requests has been withheld pending results of tests which are in progress and justification of the requested equipment.

Renovation of available buildings at the El Centro Joint Service Parachute Facility is required to support the APOLLO Earth Recovery System developmental and qualification tests. The Air Force's commitment of a C-133A aircraft to support the qualification tests has been obtained. Douglas Aircraft Company will modify the aircraft as necessary to support these tests.

AEDC facilities at Tullahoma, Tennessee, are being scheduled for use in the development of the APOLLO Reaction Control and Propulsion Systems. The use of the Mark I altitude chamber for environmental tests of the Command Module and Service Module is also planned at this time.

The design of the Integrated Mission Control Center at the MSC is presently underway. The design is being done under the direction of the Corps of Engineers with construction to start in November 1962.

Preliminary facility requirements at Cape Canaveral have been determined with A & E design criteria nearly completed for some of the facilities and some site preparation started.

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[redacted]

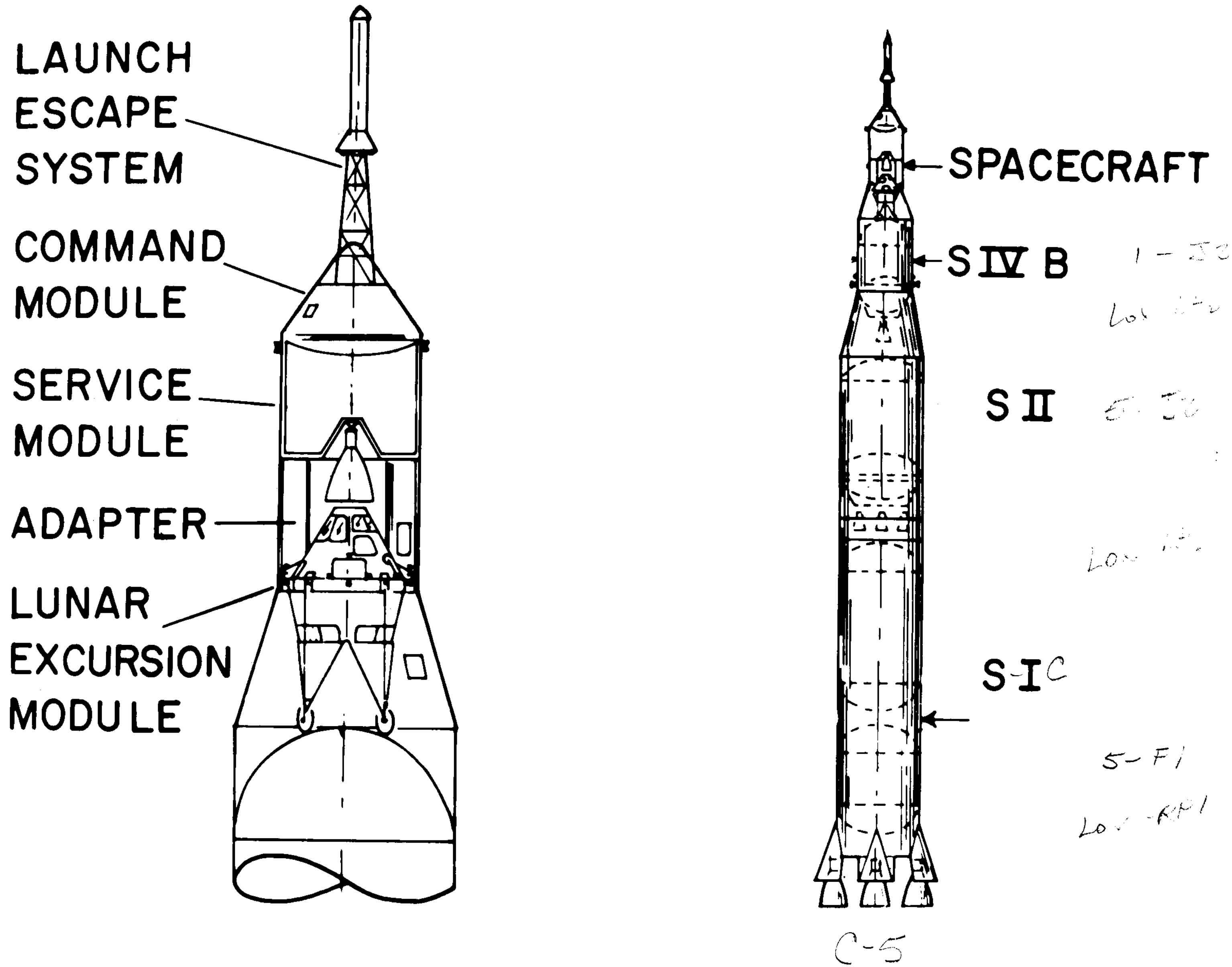

Figure 1. Apollo Space Vehicle Configuration

[redacted]

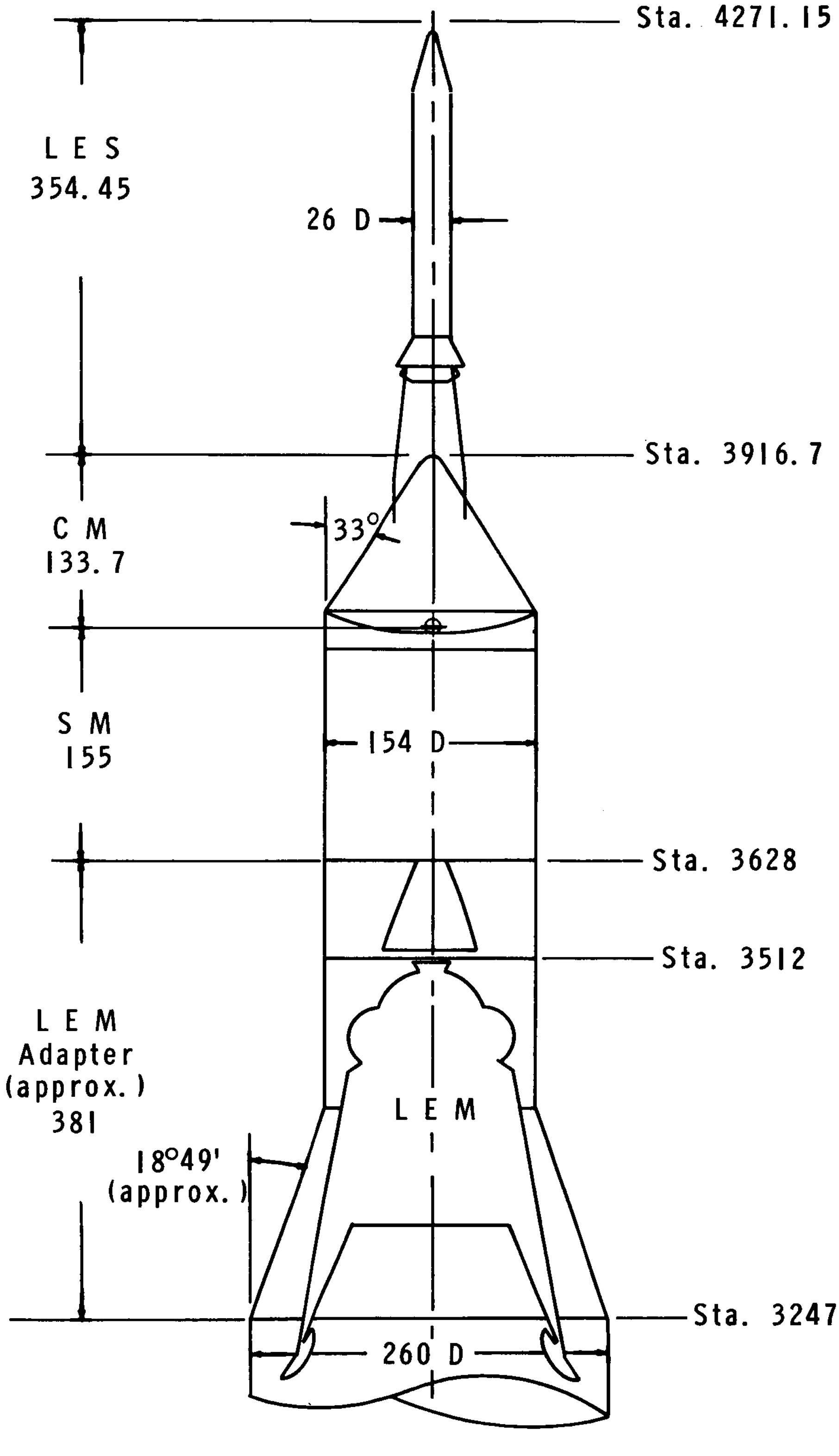

Figure 2. Apollo Spacecraft Configuration

[redacted]

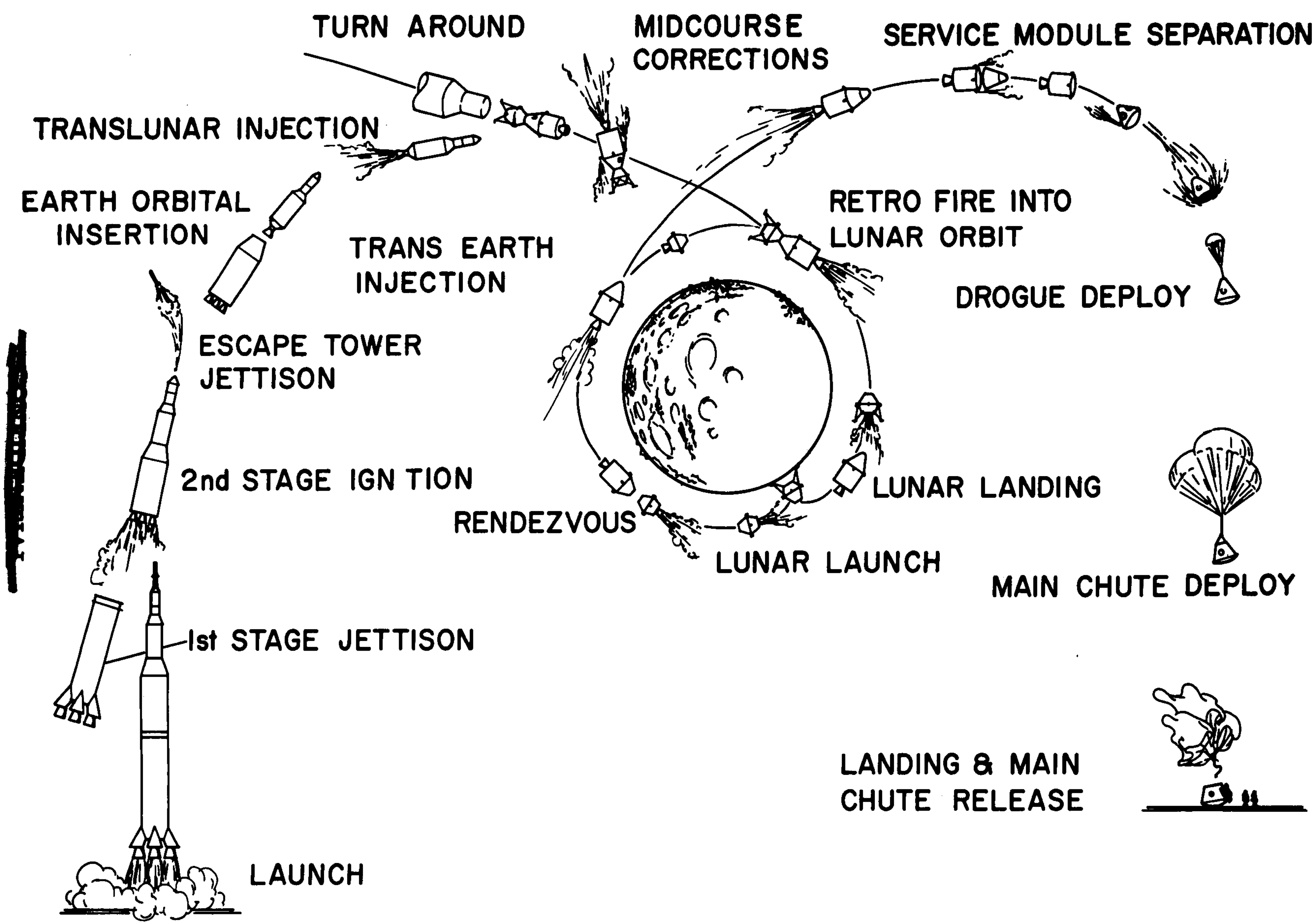

Figure 3. Typical Mission Sequence

[redacted]

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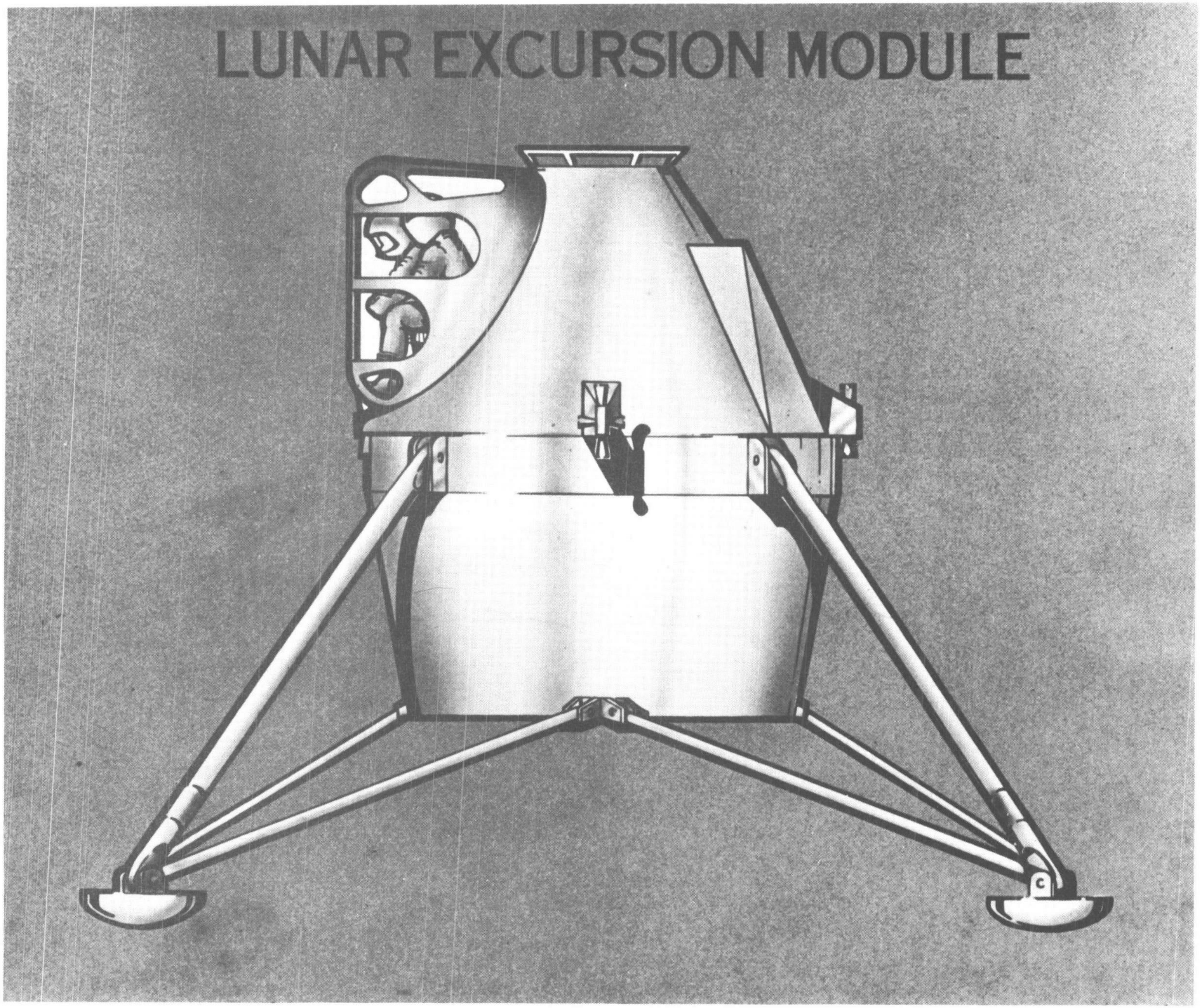

Figure 4. Lunar Excursion Module Conception

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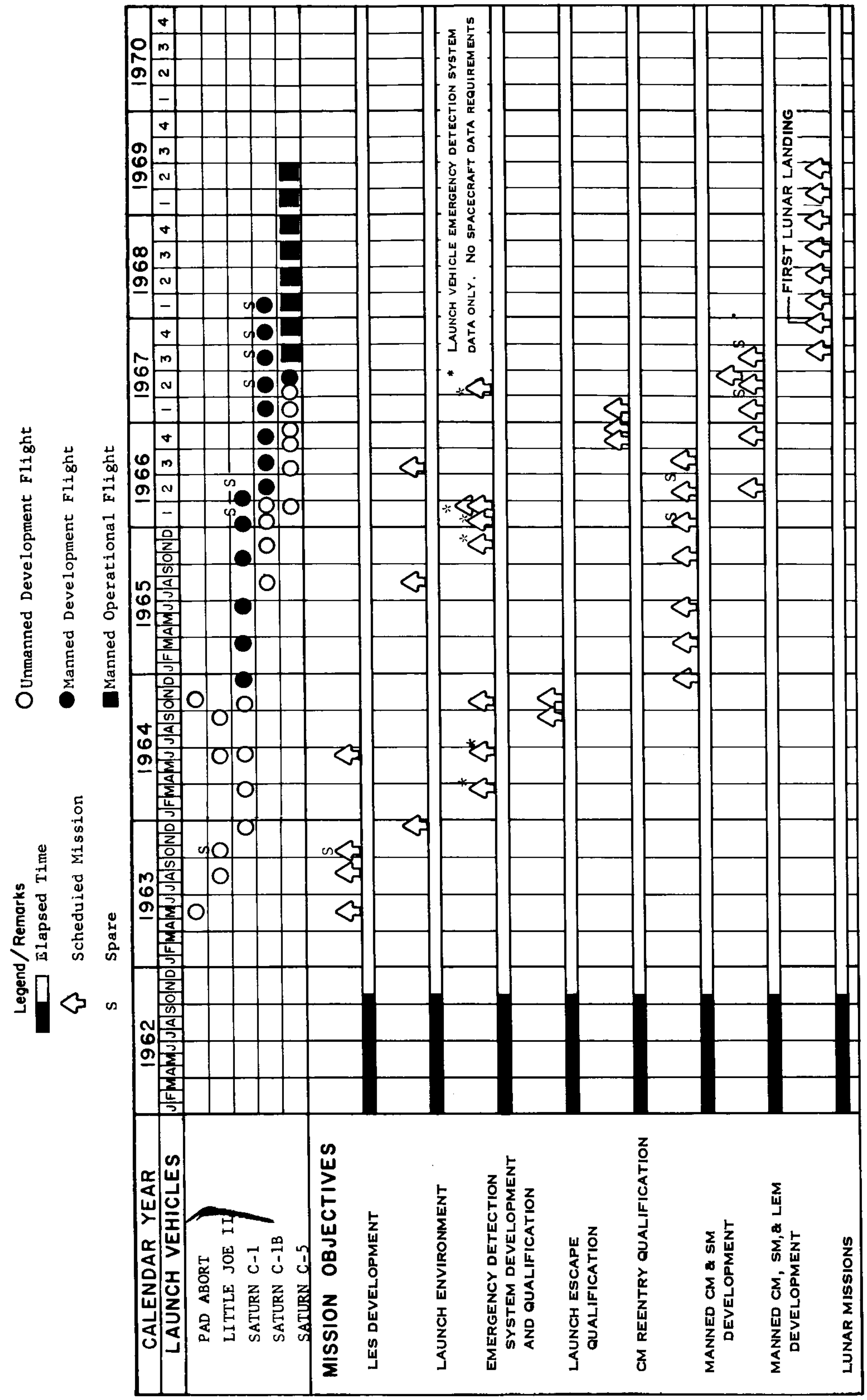

Figure 5. Project Apollo Mission Schedule

[redacted]

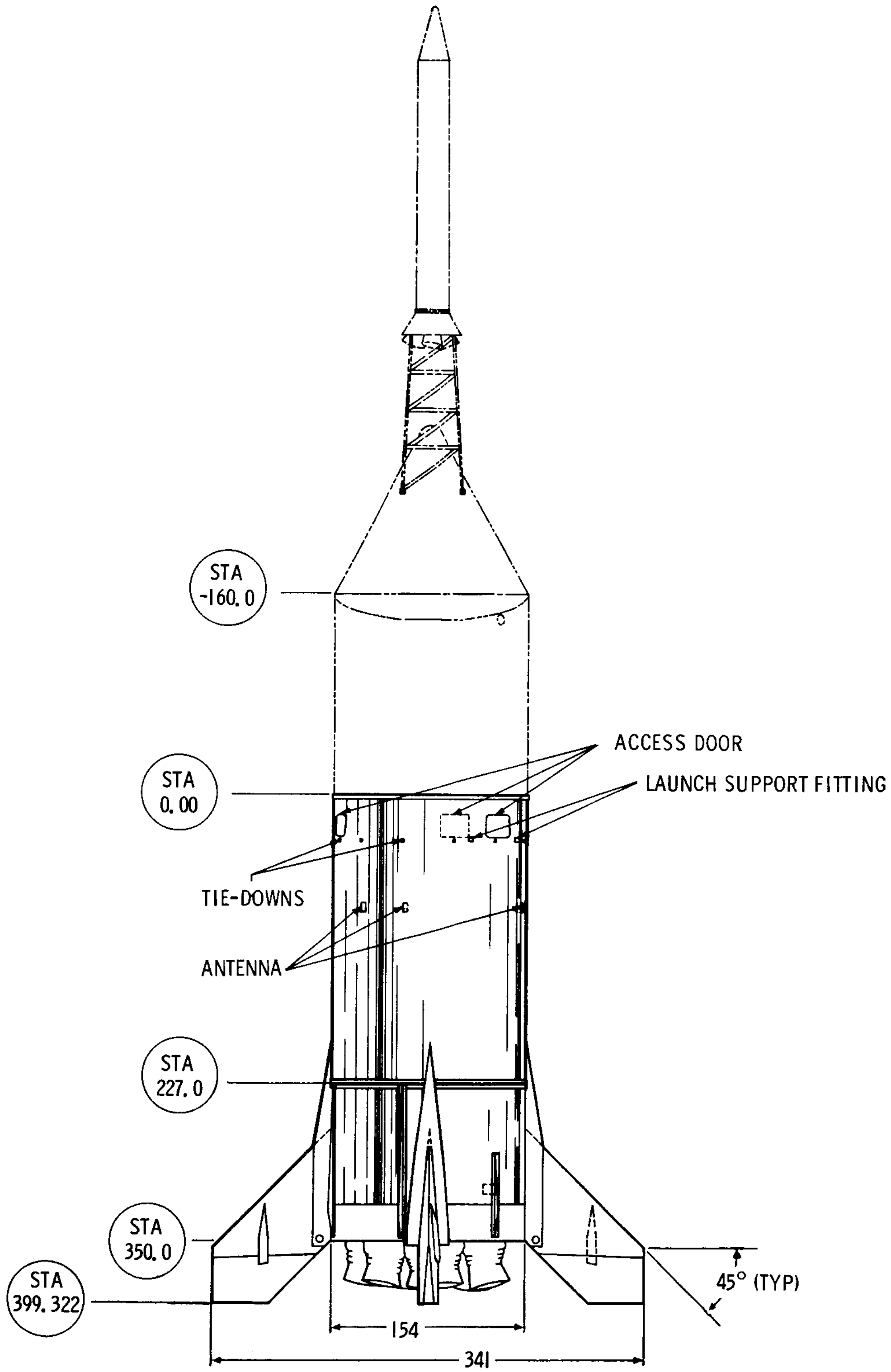

Figure 6. Little Joe II Launch Vehicle Configuration